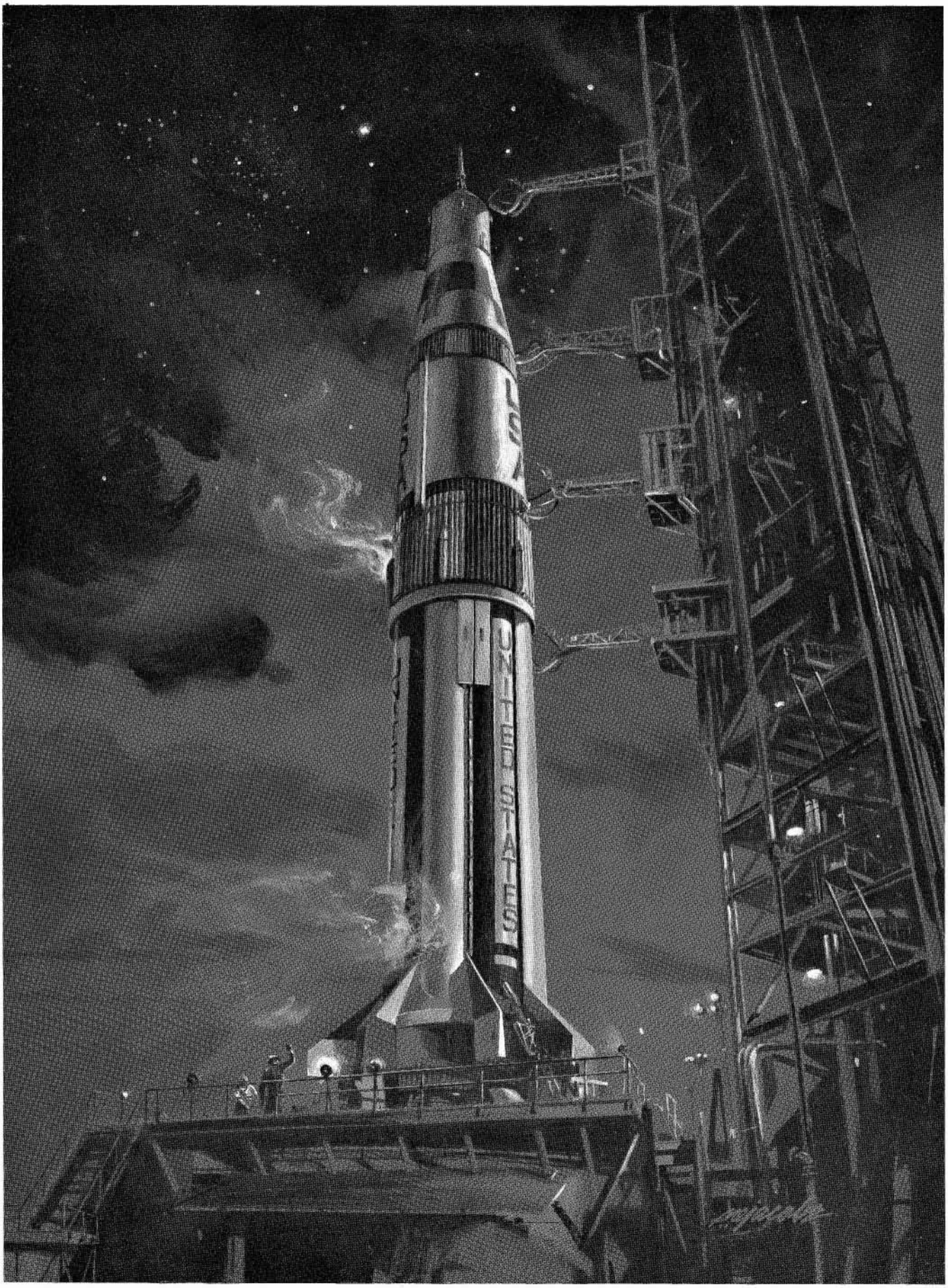
UNITED STATES

university of alabama in huntsville
saturn history
9 / 1 / 68

SATURN HISTORY DOCUMENT
University Of Alabama Research Institute
History Of Sience & Technology Group
Date ---------- Doc. No. ----------

X III.1

SATURN IB NEWS REFERENCE

DECEMBER 1965

CHANGED SEPTEMBER 1968

National Aeronautics and Space Administration
George C. Marshall Space Flight Center
John F. Kennedy Space Center

Chrysler Corporation
Space Division

McDonnell Douglas Astronautics Company

International Business Machines Corporation
Federal Systems Division

Rocketdyne
A Division of North American Rockwell Corporation

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FOREWORD

This volume has been prepared by the four Saturn IB major contractors; Chrysler, McDonnell Douglas, Rocketdyne Division of North American Rockwell, and IBM in cooperation with the National Aeronautics and Space Administration.

It is designed to serve as an aid to newsmen in present and future coverage of the Saturn IB in its role in the Saturn/Apollo Program and as a general purpose large launch vehicle. Every effort has been made to present a comprehensive overall view of the vehicle and its capabilities, supported by detailed information on the individual stages and all major systems and subsystems.

All photographs and illustrations in the book are available for general publication. The first letter(s) in each photo number is a code identifying the contractor holding that negative; CC for Chrysler, R for Rocketdyne, D for Douglas; I for International Business Machines, and N for NASA. Prints may be ordered by number by writing to the company indicated by the code.

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John F. Kennedy Space Center, Florida 32920
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THE SATURN FAMILY

INTRODUCTION TO THE SATURN PROGRAM

A definite need to loft large payloads into orbit was foreseen by the Wernher von Braun organization even before the United States orbited its first artificial satellite on January 31, 1958. Initial planning for launch vehicles having payloads of 20,000 to 40,000 pounds for orbital missions, or payloads of 6,000 to 12,000 pounds for escape missions was started in April 1957. The von Braun group, then working with the Army Ballistic Missile Agency (ABMA), submitted a "Proposal for a National Integrated Missile and Space Vehicle Development Program" to the Department of Defense in December 1957. The proposal indicated a need for a booster in the 1.5 million pound thrust class. After studies concluded that a clustered booster of 1.5 million pounds thrust was feasible, on August 15, 1958, representatives of the Advanced Research Projects Agency (ARPA) ordered the beginning of a research and development project. This project evolved into the Saturn Program.

The initial objective of the research and development program was to prove that the engine clustering technique, using existing hardware, could furnish large amounts of thrust. This was demonstrated by building and testing a single non-flight stage at Redstone Arsenal, Alabama.

Studies also showed that liquid oxygen and fuel tanks, previously developed for the Redstone and Jupiter missiles, could be modified and used for the proposed booster. It was also determined that the existing S-3D engine used on the Thor and Jupiter missile could be modified to produce an increased thrust of 188,000 pounds. Rocketdyne, a division of North American Aviation, Inc., received a contract to uprate the Thor-Jupiter engine. After redesign, simplification, and modification, the engine was identified as the H-1 engine. Initially, the thrust of the H-1 engine was 165,000 pounds; at the present time it is 200,000 pounds; and in the future it will be 205,000 pounds.

Concurrent with H-1 engine development, studies were conducted to determine the feasibility of producing a large single-chamber rocket engine capable of producing very high thrust. From these advanced studies, the 1.5 million pound thrust F-1 engine was conceived, and subsequently used as the power plant for the later Saturn boosters.

In October 1958, ARPA changed from a ground test program for proving the engine clustering concept to a program requiring the development of a reliable, high-performance booster which would serve as the first stage of a multistage vehicle capable of performing advanced space missions. This vehicle was tentatively identified as Juno V. The research group also initiated a complete vehicle study so that selection and development of an upper stage could begin.

Early in 1959, the Juno V designation was changed to Saturn, a name suggested by the relationship of the planet Saturn to the planet Jupiter. As Saturn is the next planet after Jupiter in the solar system, the Saturn rocket was the next von Braun group project following the completion of the Jupiter missile development.

Late in 1959, two decisions of far-reaching significance were made:

1. The Department of Defense decided that it had no immediate use for a large rocket, and in view of the emerging national space program, turned the Saturn project over to the newly-formed National Aeronautics and Space Administration (NASA).
2. NASA formed a Saturn Vehicle Evaluation Committee (the Silverstein Committee) composed of NASA and Defense officials. The committee recommended that all upper stages of Saturn vehicles be powered by the high energy propellant combination of hydrogen and oxygen, and that a new hydrogen engine be developed.

Development of the J-2 engine and three upper stages resulted from these decisions. A building block approach to a series of successively larger vehicles was outlined by NASA early in 1960. The first, Saturn I, was to be a three-stage vehicle, ten of which were planned. Subsequently, by increasing the thrust of the second stage, the planned third stage was eliminated.

During 1960, Douglas Aircraft Company, Inc. was selected to build the second stage of Saturn I. Designated as the S-IV stage, it was powered by six Pratt and Whitney RL10A-3 engines. Rocketdyne was chosen to develop the new hydrogen fueled J-2 engine to be used in later vehicles of the Saturn program.

In the spring of 1960, successful static firings of the S-I stage took place to verify the clustered engine technique as a basic consideration for still larger vehicles.

At midyear 1960, the von Braun development group handling Saturn work was formally trans-

ferred to NASA, and became the nucleus of the newly created George C. Marshall Space Flight Center.

Chrysler Corporation was awarded a contract in June 1961 to qualify and test S-I stage engine, hydraulic, mechanical, and structural components.

In May 1961, the late President Kennedy's challenge to the nation to place astronauts on the moon in this decade created an immediate necessity for a launch vehicle considerably larger than the Saturn I.

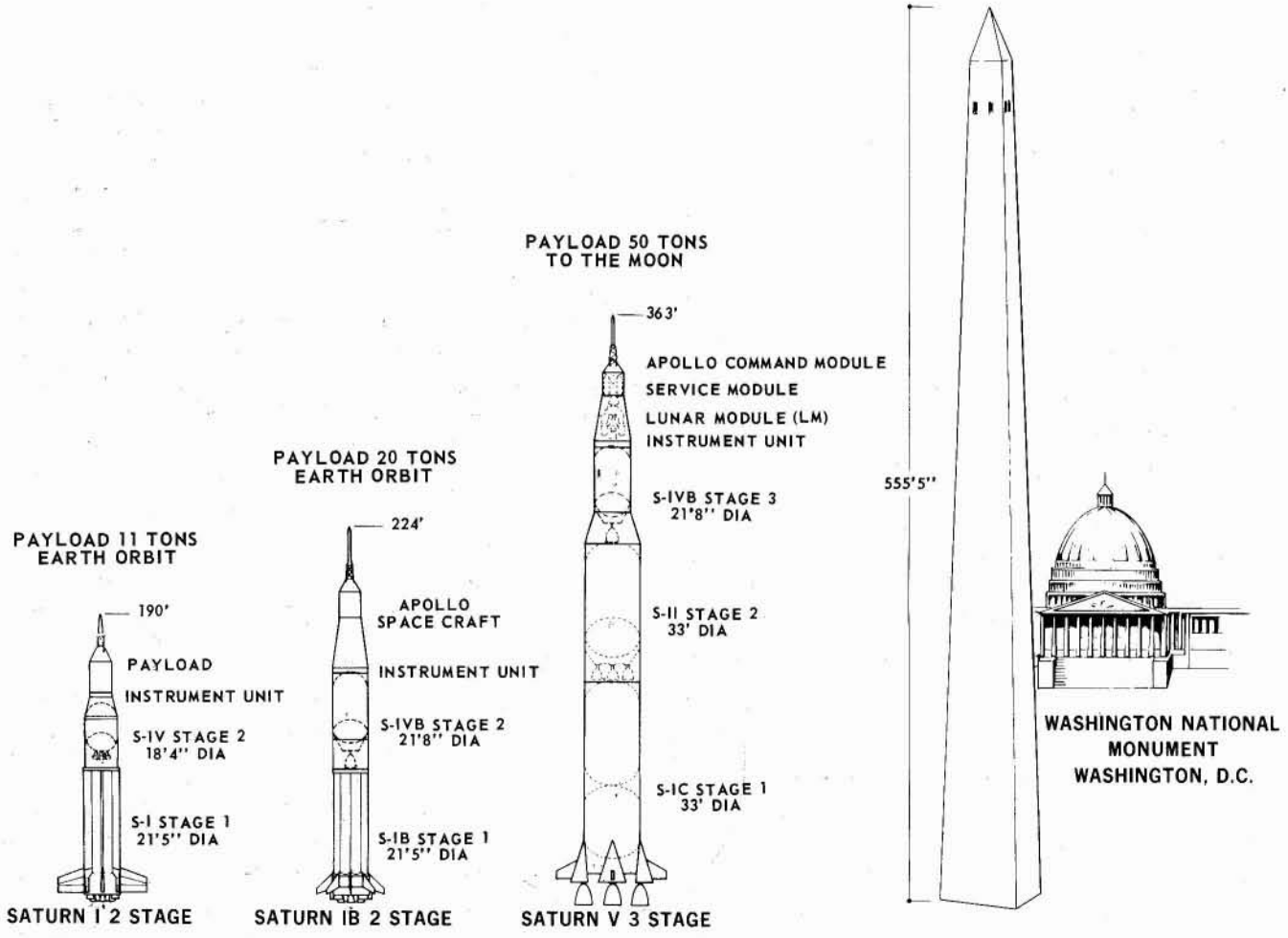
Following more than six months of intensive study, NASA announced in January 1962 that the next Saturn vehicle would be the Saturn V with a ground stage thrust of 7.5 million pounds — five times that of Saturn I. It would be capable of placing more than 120 tons into earth orbit. This larger vehicle would have two new upper stages: a new S-II second stage and S-IVB third stage, both utilizing the new J-2 engines. The S-IVB stage would be an adaptation of the S-IV stage already developed for the Saturn I.

As the project to land Americans on the moon was studied, it was determined that the nation would not build one huge rocket for a direct flight from earth to the moon's surface. Instead, two rendezvous approaches were studied:

1. Bringing together two Saturn V payloads in earth orbit to form a moon ship, and then proceed to a moon landing.
2. Launching a single Saturn payload into lunar orbit, from which a small landing craft would be dispatched to the moon's surface, and later rendezvous with the mother ship still in lunar orbit.

Both the earth orbital rendezvous and lunar orbital rendezvous missions would use the Saturn V launch vehicle.

Finally, in July 1962, it was announced that on the basis of cost, safety, and time, the lunar orbit rendezvous method was favored. This decision entailed the use of still another launch vehicle with a capability between the Saturn I



Saturn Comparison

D-PB-5A

and the Saturn V to test the complete Apollo spacecraft in earth orbit as soon as possible. The new vehicle was identified as the Saturn IB, and would be comprised of a modified Saturn I first stage (S-IB) and Saturn V third stage (S-IVB).

The Saturn IB permitted flight testing of the complete spacecraft about one year earlier than would have been possible had NASA waited for availability of the Saturn V.

As the plan stood, the Saturn I would be used to place early, unmanned Apollo command and service modules into earth orbit; the Saturn IB would launch those two modules plus a moon landing craft, lunar excursion module, into earth orbit for astronaut training and rendezvous practice; and the Saturn V would provide power for the lunar landing. Thus, by marrying the elements of Saturn I and Saturn V to form the Saturn IB, manned earth orbital rendezvous flights could begin a year earlier without the expense of a completely new development program.

Saturn I

While plans for the lunar mission were progressing, the Saturn I project made history. On October 27, 1961, the first Saturn I booster was flight-tested successfully from Kennedy Space Center (KSC). The first flight booster with dummy upper stages was called SA-1. This vehicle was followed by successful flights of SA-2 on April 25, 1962, SA-3 on November 16, 1962, and SA-4 on March 28, 1963.

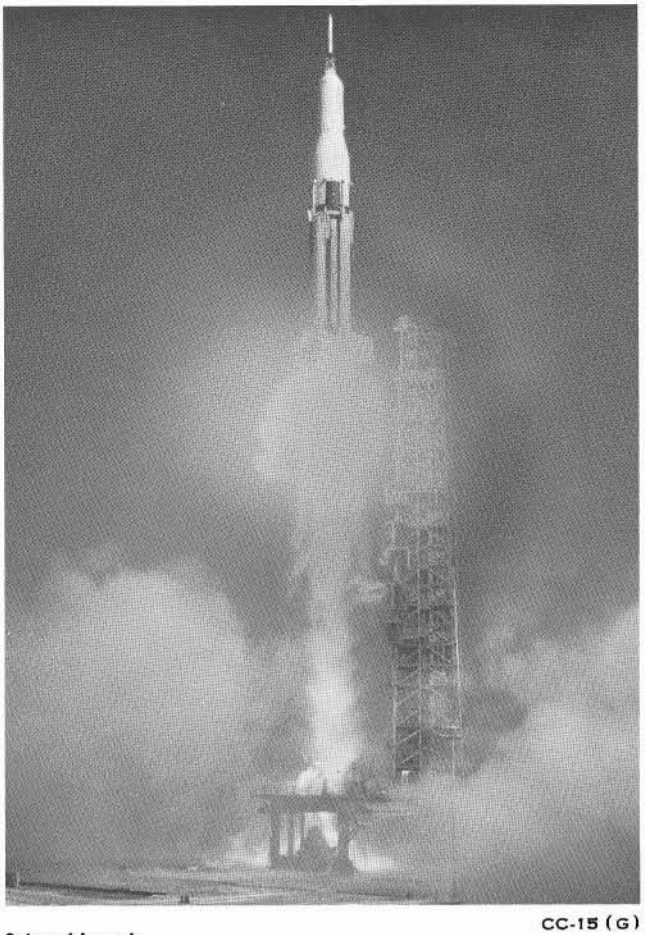
The SA-5 vehicle, combining the first stage S-I with an S-IV stage, was successfully launched on January 29, 1964, with both stages functioning perfectly to place a 37,700 pound payload into earth orbit. SA-6, launched on May 28, 1964, and SA-7, launched on September 18, 1964, each placed unmanned "boilerplate" configurations of Apollo spacecraft into earth orbit.

SA-9, launched on February 19, 1965, was the first Saturn I vehicle to launch a Pegasus meteoroid technology satellite into earth orbit to measure the amount and size of space particles.

The SA-8 and SA-10 Saturn I vehicles were successfully launched from KSC on May 25, 1965, and July 30, 1965, respectively, to complete the test and launch program with an unprecedented 100 per cent record of success.

The Saturn IB

Based upon the technology of the Saturn I program, the Saturn IB uses the S-IB first stage which is a modified version of the S-I stage, together with the S-IVB second stage, an upgraded version of the S-IV stage, and an Instrument Unit originally designed for the future Saturn V launch vehicle.



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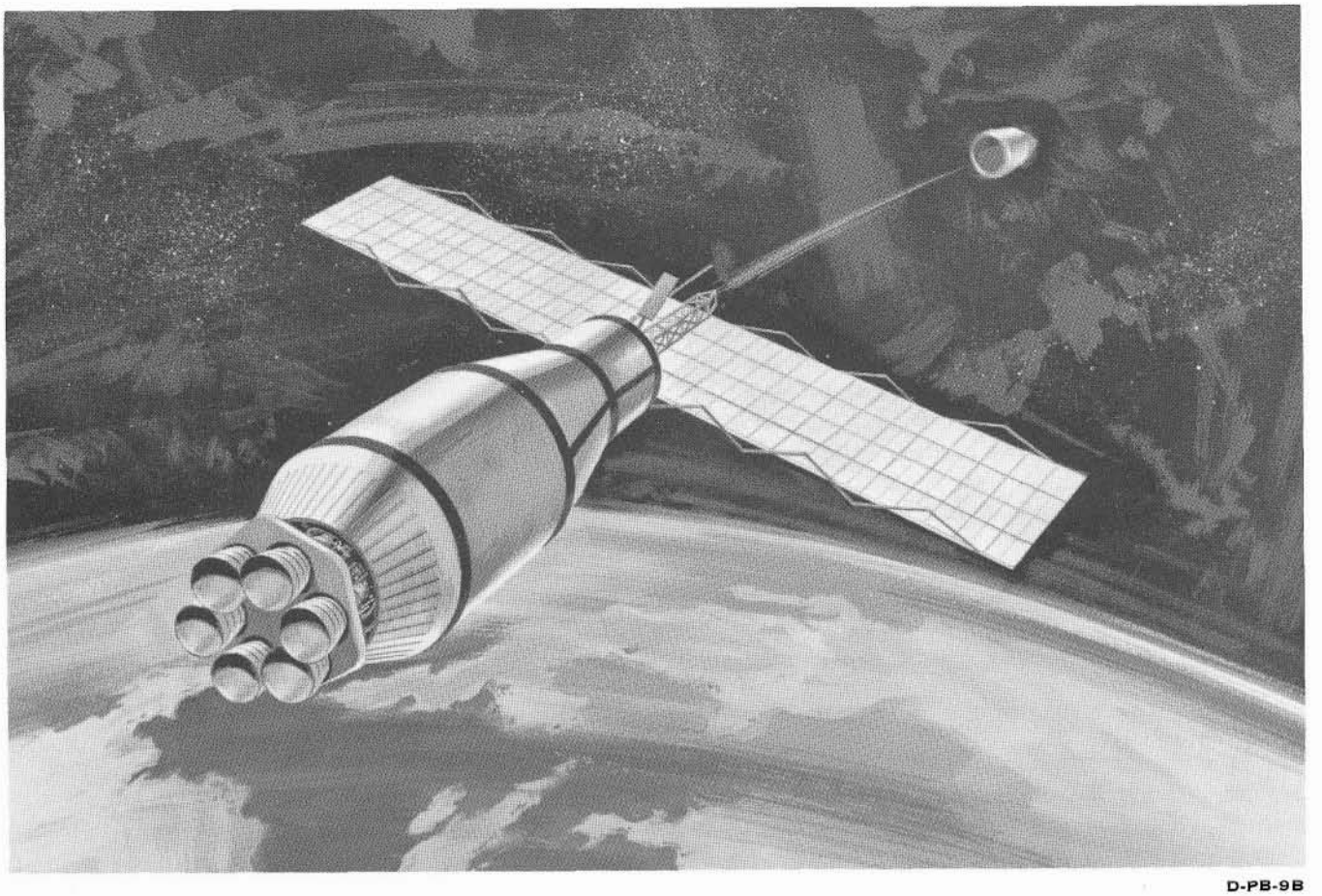
Saturn I Launch

The S-I first stage was redesigned in several areas by NASA and Chrysler for its expanded role as the Saturn IB booster. Basically, it retained the same shape and size, but required some modification for mating with the S-IVB stage, which has a greater diameter and weight than the S-IV stage.

Stage weight was cut by more than 20,000 pounds to increase payload capacity. This reduction was accomplished by a new fin design, removing hydrogen vent pipes and brackets unnecessary to the new design, resizing machined parts in the tail section assembly, redesigning the spider beam, and modifiying the propellant tanks. The Rocketdyne H-1 engine was uprated to 200,000 pounds of thrust, compared with 188,000 pounds of thrust for each engine in the Saturn I, Block II. The engines will be uprated again to 205,000 pounds beginning with the SA-206.

Early development of the S-IVB stage to meet the schedule of the Saturn IB was possible by drawing on the technology gained from Douglas development of the S-IV stage for the Saturn I.



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Modern-Day Pegasus in flight high above the earth, with the Saturn I/S-IV stage attached to the meteoroid detection payload, while the unmanned Apollo spacecraft soars ahead in the same orbit after being separated from the S-IV stage by a spring mechanism. The orbiting vehicle, including the S-IV and Apollo, is about 70 feet long, and Pegasus "wings" measure 96 feet across.

The 200,000 pound thrust Rocketdyne engine more than doubled the S-IV stage thrust capability. Development of the J-2 engine also drew heavily upon large-engine technology experience in hydrogen pumping acquired under advanced engine group (AEG) sponsored programs.

The Instrument Unit used on the Saturn IB is nearly identical to that used on the Saturn V. Equipment used in the Saturn I Instrument Unit program was intended to test the concepts for design of the Saturn V Instrument Unit. There are a few carryover components; however, later Saturn I vehicles used an inertial platform and control computer similar in design and operation to that being used in the Saturn IB.

The guidance computer used in the early Saturn I vehicle was an adaptation of a computer developed by International Business Machines for use in Titan II. For the Saturn IB, it is replaced by an IBM computer of completely new design which incorporates the added flexibility and extreme reliability necessary to carry out the intended Saturn IB missions.

The equipment used in the Instrument Unit represents a unique blend of old and new technologies. Due to military requirements, early missile programs were concerned with accurate delivery of inanimate payloads after a relatively short period of powered flight. Automatic control systems were the prime requirements to provide guidance and control for these types of military vehicles.

The addition of man as an extremely important consideration in Saturn IB design meant that new systems had to be developed, while skillfully adapting the best features of older systems for longer durations, varied objectives, and an overriding concern for the safety of the human passenger.

Saturn V

Saturn V, third and largest member of the Saturn family, is a three-stage vehicle capable of sending a 50-ton payload to the moon, or boosting as much as 125 tons into low earth orbit. As the Apollo lunar launch vehicle, Saturn V will stand 364 feet high, and when fully fueled will weigh over six million pounds.

The S-IC first stage will be powered by five Rocketdyne F-1 engines, each having 1.5 million pounds of thrust, for a total of 7.5 million pounds. This stage will be 33 feet in diameter and 138 feet long, and will use liquid oxygen and RP-1 (kerosene) as propellants.

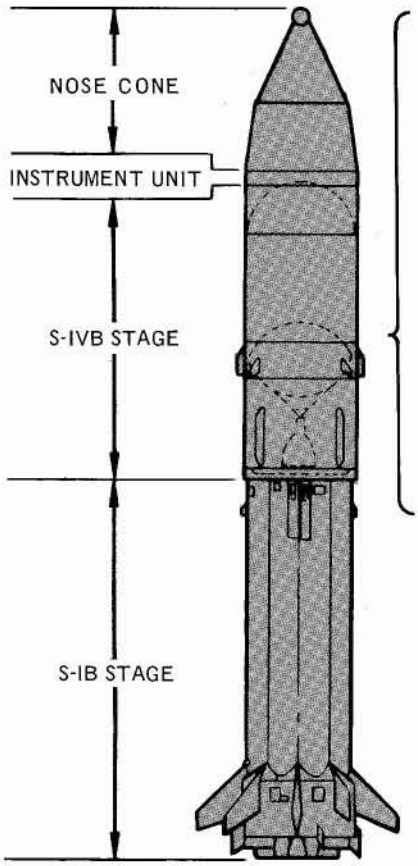
The S-II second stage will also be 33 feet in diameter, with a total length of 81.5 feet, and will use liquid oxygen and liquid hydrogen propellants. This stage will have a total thrust of one million pounds provided by five 200,000 pound thrust Rocketdyne J-2 engines.

The S-IVB third stage, which also serves as the upper stage of the Saturn IB, will be 21.7 feet in diameter and 58.4 feet long, and the hydrogen-fueled J-2 engine will provide 200,000 pounds of thrust. The J-2 engine will be modified to provide an in-space restart capability to meet requirements of the Saturn V lunar launch mission.

The Saturn V Instrument Unit built by IBM is nearly identical to that used on the Saturn IB, with no change in physical dimensions or internal systems; however, minor modifications in instrumentation will be made to meet the Saturn V mission requirements.

SATURN IB FACT SHEET

VEHICLE AS-203 MAJOR MODIFICATIONS TO ACCOMMODATE LIQUID HYDROGEN FUEL EXPERIMENT



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MODIFICATIONS

TV Camera Equipment
Telemetry
Stage Power
Continuous Vent System
Insulation
Tank Baffling & Deflector
Instrumentation Probe
Antivortex Screen
Sequencer
Repressurization System
Ullage Thrust System

NOTE: Detailed modification data are contained in S-IVB STAGE MODIFICATIONS, Addendum 1.

in NOTES

MISSION PROFILE

The primary mission of the Apollo/Saturn (AS) 203 vehicle is to study the behavior of liquid hydrogen in orbit. The S-IVB stage serves as the second stage and experimental payload of AS-203. The stage is configurated to perform the major in-orbit functions of the S-IVB stage of a Saturn V vehicle.

The S-IVB stage will be launched into a 100-mile orbit, with approximately 19,000 pounds (32,590 gallons) of liquid hydrogen remaining in its fuel tank after orbit is achieved. This is approximately one-half of the total fuel load.

To accommodate the primary mission, two subsystems are installed which are designed to simulate the S-IVB stage of Saturn V: hydrogen propulsive vent system and fuel tank repressurization system. A gaseous oxygen propulsive vent is also added to simulate ullage control thrustors. Other modifications provided are to support experiments and studies of liquid hydrogen mass under zero G or low gravity conditions.

Information obtained from studies of the AS-203 flight will help confirm the design of systems for the S-IVB stage of the Saturn V related to orbital coast and restart capabilities. It will also provide an opportunity for evaluation of possible problem areas in restarting the S-IVB in space well before the first Saturn V is launched.

PHYSICAL CHARACTERISTICS

	Diameter	Height	Weight
OVERALL VEHICLE		173 ft	126,200 lb (dry) 1,193,700 lb (approx) (total liftoff)
S-IB STAGE	21.4 ft	80.3 ft	86,200 lb (dry)
S-IVB STAGE	21.7 ft	58.4 ft	31,700 lb (dry)
INSTRUMENT UNIT	21.7 ft	3.00 ft	4,600 lb
NOSE CONE	21.7 ft (at max)	31.3 ft	3,700 lb

PROPULSION SYSTEMS

S-IB STAGE — Eight bipropellant H-1 engines developing 1,600,000 lb thrust
RP-1 Fuel — 43,000 gal (279,000 lb)
LOX 68,200 gal (632,500 lb)

S-IVB STAGE — One bipropellant J-2 engine developing 200,000 lb thrust
LH_2 — 64,000 gal (38,000 lb)
LOX — 12,420 gal (118,000 lb)

CAPABILITY

S-IB STAGE — Operates approximately 2.5 minutes to reach an altitude of 42 miles at burnout.
S-IVB STAGE — Operates approximately 294 seconds to achieve orbital speed and altitude.
INSTRUMENT UNIT — Supplies electronic commands for steering, engine ignition and cutoff, and staging operations.
PAYLOAD — 19,000 lb LH_2 in earth orbit.

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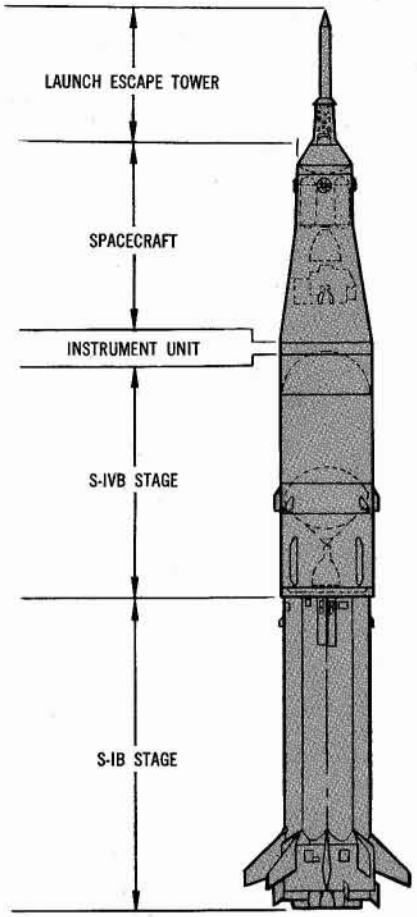
SATURN IB FACT SHEET

VEHICLE AS-202



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PHYSICAL CHARACTERISTICS

	Diameter	Height	Weight
OVERALL VEHICLE		224 ft	153,200 lb (dry) 1,312,000 lb (approx) (total liftoff)
S-IB STAGE	21.4 ft	80.3 ft	91,500 lb (dry)
S-IVB STAGE	21.7 ft	58.4 ft	29,650 lb* (dry)
INSTRUMENT UNIT	21.7 ft	3.00 ft	4,500 lb
SPACECRAFT	21.7 ft (at max)	52.7 ft	48,400 lb (fueled)
LAUNCH ESCAPE TOWER	2.2 ft	33.3 ft	8,500 lb

*Including 6,400 lb aft interstage

PROPULSION SYSTEMS

S-IB STAGE — Eight bipropellant H-1 engines developing 1,600,000 lb thrust
RP-1 Fuel — 42,000 gal (270,500 lb)
LOX — 64,000 gal (611,000 lb)

S-IVB STAGE — One bipropellant J-2 engine developing 200,000 lb thrust
LH_2 — 62,500 gal (36,000 lb)
LOX — 20,000 gal (191,000 lb)

CAPABILITY

S-IB STAGE — Operates approximately 2.5 minutes to reach an altitude of approximately 37 miles at burnout.

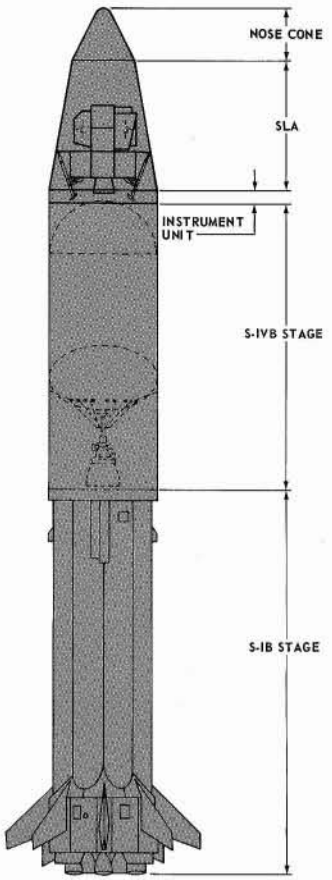
S-IVB STAGE — Operates approximately 7.5 minutes to achieve orbital speed and altitude.

INSTRUMENT UNIT — Supplies electronic commands for steering, engine ignition and cutoff, and staging operations.

PAYLOAD — 48,385 lb on sub-orbital trajectory.

SATURN IB FACT SHEET

APOLLO 5 (VEHICLE AS-204) FLIGHT EXPERIMENT



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MISSION PROFILE

The Apollo 5 mission (vehicle designated Apollo/Saturn 204) will be the first space test of the Apollo lunar module (LM) destined to take astronauts to the lunar surface and return them to the Apollo command spacecraft in lunar orbit.

Purpose of this flight is to verify the unmanned LM's propulsion systems and to further verify the launch vehicle's performance for future manned space flight. Also planned for this flight will be the second (S-IVB) stage propellant dump experiment after LM separation in preparation for later Saturn flights.

The Uprated Saturn I will be launched from complex 37B at the NASA-Kennedy Space Center to an elliptical orbital altitude of approximately 102 by 138 statute miles. A payload consisting of the S-IVB, instrument unit, SLA (spacecraft LM adapter) panels, Lunar Module 1, and nose cone will go into orbit. Soon after orbit is achieved the nose cone will be jettisoned and 10 minutes later the SLA panels will be deployed. About 30 minutes later the LM will be separated and checked out in orbit. The S-IVB and instrument unit will maintain attitude control for about three orbits.

PHYSICAL CHARACTERISTICS*	Diameter	Height	Weight
OVERALL VEHICLE		180.9 ft	156,600 lb (dry) 1,285,000 lb (approx) (total liftoff)
S-IB STAGE	21.4 ft	80.3 ft	85,317 lb (dry)
S-IVB STAGE	21.7 ft	58.4 ft	30,300 lb**
INSTRUMENT UNIT	21.7 ft	3.00 ft	4,600 lb
SPACECRAFT L/M ADAPTER	21.7 ft (at max)	28.0	3,950 lb
NOSE CONE	12.8 ft (at max)	11.3 ft	1,067 lb
LUNAR MODULE (inside L/M adapter)			31,325 lb

*Data for AS-204 only; may vary with nominal data cited on following pages.

**Including 6,654 lb aft interstage

PROPULSION SYSTEMS

S-IB STAGE — Eight bipropellant H-1 engines developing 1,600,000 lb thrust
RP-1 Fuel — 279,065 lb (41,693 gal)
LOX — 630,491 lb (66,899 gal)

S-IVB STAGE — One bipropellant J-2 engine developing 225,000 lb thrust (maximum)
LH_2 — 35,000 lb (58,920 gal)
LOX — 189,400 lb (19,850 gal)

CAPABILITY

S-IB STAGE — Operates approximately 2.4 minutes to reach an altitude of 39.6 miles at burnout.

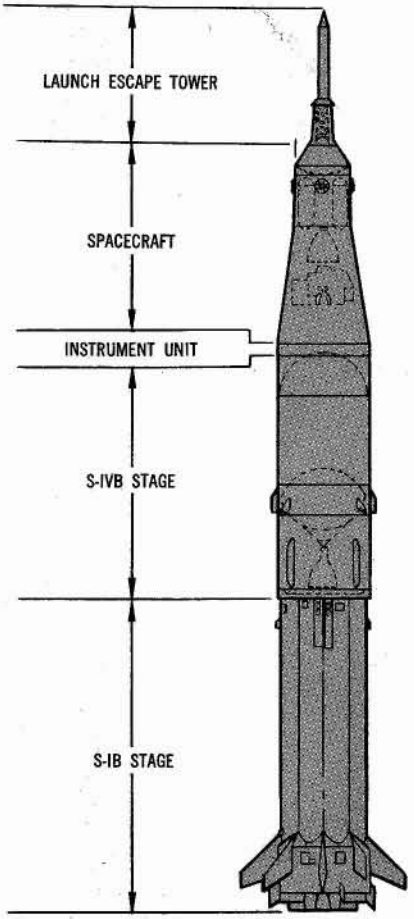
S-IVB — Operates approximately 7.5 minutes to achieve orbital speed and altitude.

INSTRUMENT UNIT — Supplies electronic commands for steering, engine ignition and cutoff, and staging operations.

PAYLOAD — 36,300 lb in earth orbit.

SATURN IB FACT SHEET

APOLLO 7 (VEHICLE AS-205) FLIGHT EXPERIMENT



D-PB-201

MISSION PROFILE

The Apollo 7 mission (vehicle designated Apollo/Saturn 205) will be the first manned space test of the Apollo command/service modules.

Purpose of this flight is to verify the spacecraft/crew operations and subsystems performance for an earth orbital mission. Primary objectives are (1) demonstrate CSM/crew performance in an orbital environment; (2) demonstrate crew/space vehicle/mission support facilities performance; (3) demonstrate adequacy of the launch vehicle attitude control system for orbital operation; (4) demonstrate CSM active rendezvous with the S-IVB/IU/SLA; and (5) demonstrate S-IVB orbital safing capability. Secondary objectives are to evaluate the S-IVB/IU orbital coast lifetime capabilty and demonstrate CSM manual attitude control of the launch vehicle in orbit.

The Saturn IB will be launched from complex 34 at the NASA-Kennedy Space Center to an elliptical orbit of approximately 138 by 173 statute miles. The S-IVB, IU, SLA (spacecraft LM adapter) and the Apollo CSM (command/service modules) will go into orbit. Total weight to be injected into orbit is approximately 66,850 pounds, including propellants. At 1 hour and 34 minutes into the flight, a 12-minute LOX dump begins, followed at 1 hour and 42 minutes by the beginning of a 47-minute cold helium dump. Manual control of the S-IVB attitude from the spacecraft begins at 2½ hours into the flight and ends 7 minutes later. The CSM separates from the S-IVB 2 hours and 55 minutes after liftoff.

PHYSICAL CHARACTERISTICS*	Diameter	Height	Weight
OVERALL VEHICLE		224 ft	153,361 (dry) 1,290,184 lb (approx) (total liftoff)
S-IB STAGE	21.4 ft	80.3 ft	84,400 lb (dry)
S-IVB STAGE	21.7 ft	58.4 ft	28,380 lb**
INSTRUMENT UNIT	21.7 ft	3.00 ft	4,280 lb
SPACECRAFT L/M ADAPTER	21.7 ft (at max)	28.0 ft	3,820 lb
APOLLO CSM	12.9 ft (at max)	34.0 ft	32,480 lb***

*Data for AS-205 only; may vary with nominal data cited on following pages.

**Including 6,478 lb aft interstage.

***Including 8,930 lb SPS tanked.

PROPULSION SYSTEMS

S-IB STAGE — Eight bipropellant H-1 engines developing 1,600,000 lb thrust
RP-1 Fuel — 277,216 lb (42,000 gals.)
LOX — 631,346 lb (67,000 gals.)

S-IVB STAGE — One bipropellant J-2 engine developing 225,000 lb thrust (maximum)
LH_2 — 37,348 lb (64,000 gals.)
LOX — 193,273 lb (20,000 gals.)

CAPABILITY

S-IB STAGE — Operates approximately 2.4 minutes to reach an altitude of 37.6 miles at burnout.

S-IVB — Operates approximately 7.5 minutes to achieve orbital speed and altitude.

INSTRUMENT UNIT — Supplies electronic commands for steering, engine ignition and cutoff, and staging operations.

PAYLOAD — 36,600 lb in earth orbit.

SATURN IB DESIGN FEATURES

VEHICLE CONCEPT

The Saturn IB launch vehicle was conceived in 1962 at the NASA Marshall Space Flight Center as the quickest, most reliable, and most economical means of providing a booster with greater payload capability than the Saturn I. The new launch vehicle would be used for earth orbital missions with the Apollo spacecraft before the Saturn V lunar launch vehicle would be available.

Development of the Saturn IB was based on a blending of existing designs for the Saturn I and the Saturn V. It uses a redesigned Saturn I booster (designated the S-IB stage), together with the S-IVB upper stage and the Instrument Unit from the Saturn V.

The concept permitted rapid development of a new vehicle. Maximum use of designs and facilities available from the earlier approved Saturn programs, saved both time and costs.

Saturn IB thus becomes a second generation of the Saturn family — the first U.S. rocket boosters developed from the start as large payload, manned space launch vehicles.

VEHICLE DESCRIPTION

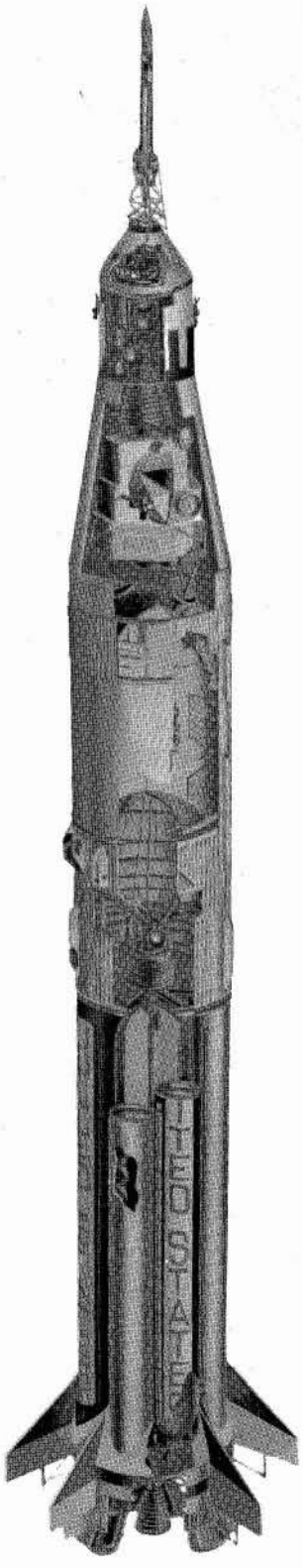
Saturn IB, including the spacecraft and tower, stands approximately 224 feet tall, and is about 21.7 feet in diameter. Total weight empty is about 76 tons, and liftoff weight fully fueled, will be approximately 650 tons.

First-stage flight is powered by eight H-1 engines generating 200,000 pounds of thrust each, for a total of 1.6 million pounds. In approximately 2.5 minutes of operation, it will burn 42,000 gallons of RP-1 fuel and 67,000 gallons of liquid oxygen, to reach an altitude of approximately 40 miles at burnout. H-1 engines for later S-IB vehicles will be uprated to 205,000 pounds of thrust each.

The S-IVB stage, with a single 200,000 pound thrust J-2 engine, burns 64,000 gallons of liquid hydrogen and 20,000 gallons of liquid oxygen in about 7.5 minutes of operation, to achieve orbital speed and altitude. Thrust of the J-2 will be uprated in later Saturn IB vehicles.

The Instrument Unit is the Saturn IB "brain" responsible for originating electronic commands for stage steering, engine ignition and cutoff, staging operations, and all primary timing signals.

Primary payload for the Saturn IB is the Apollo spacecraft which is being developed by NASA



Saturn IB Launch Vehicle

D-PB-200

for manned flights to the moon. It will be carried atop the Instrument Unit to complete the vehicle's launch configuration.

MISSION

Tests of the Apollo spacecraft in both manned and unmanned flights are the initial missions assigned to the Saturn IB launch vehicle.

In keeping with the "all-up" philosophy of flight test, the first launches have been made with the first and second stages and the instrument unit fully active, and have carried live payloads. On the second flight, the S-IVB stage itself was the payload, serving as an orbital test bed for experiments related to the behavior of liquid hydrogen under low-gravity conditions.

In the Apollo Applications Program, an S-IVB stage will become a manned Saturn I Workshop, with its liquid hydrogen tank converted into living and working quarters for a three-man crew after the fuel is depleted in achieving Earth orbit.

DEVELOPMENT HIGHLIGHTS

Because of NASA's original determination to make maximum use of technology and equipment already existing or under design, Saturn IB was brought to full development in less than four years after the initial go-ahead decision.

In that time, Marshall Space Flight Center and Chrysler Corporation have completed necessary modifications and uprating on the S-IB stage; Douglas has developed the S-IVB stage for the Saturn IB and accelerated production and testing to meet the launch schedule; MSFC and IBM Federal Systems Division have done the same in adapting the Saturn V Instrument Unit for Saturn IB; and Rocketdyne has uprated the H-1 engines for the S-IB first stage, and stepped up development and production of the J-2 engine for the S-IVB second stage.

The first S-IB booster was test fired at MSFC on April 1, 1965, and subsequently delivered to Kennedy Space Center, Florida, in mid-August.

The second stage for the first Saturn IB flight vehicle was acceptance fired at the Douglas Sacramento Test Center on August 8, 1965, and delivered to KSC on September 19.

The Instrument Unit for the Saturn IB was delivered to KSC on October 20, and mating of the IU and the rocket stages was completed at Launch Complex 34 on October 25. The first Saturn IB flight vehicle was thus completed just 39 months after the initial NASA decision to proceed with its development.

TECHNICAL ADVANCES

Automatic Checkout

Saturn IB is the first major space launch vehicle to employ completely automated, computer-controlled checkout systems for each of its stages. The capability was initially operational on the S-IVB and the Instrument Unit, and on the S-IB stage for all vehicles after the fourth launch.

The Automatic Checkout System (ACS) uses a carefully detailed computer program and associated electronic equipment to perform a complete countdown checkout of each stage and all its various systems, subsystems, and components.

With electronic speed, it moves through a more thorough and more reliable countdown than is humanly possible. Yet the system permits test engineers to monitor every step of the operation, and to over-ride the computer's functions if necessary.

With electronic signals, the computer tests each item on the extensive check-list programmed into its memory. It compares the response with the result it is programmed to expect.

On receiving a proper response, the computer automatically moves ahead to the next test. But if any tested component fails to respond correctly, the computer automatically indicates the failure at the control console. The machine can pin-point the malfunction for the test conductor. It can also automatically indicate ways to double-check a questionable response, in order to further define any difficulty.

The computer system is used for the final factory checkout of each S-IVB and Instrument Unit. It is used in pre-firing checkouts of the S-IVB before the acceptance test; performs the final countdown for the static firing, and controls the actual firing; and it is used again for post-test checkouts.

At Kennedy Space Center, pre-launch checkout and actual launch control functions for the entire Saturn IB also will be computer operated.

The automatic control technology developed for the Saturn program shows promise of significant technical "fall-out" for application in many commercial and industrial applications where rapid, accurate testing of complex equipment is necessary.

J-2 Engine

The J-2 engine which powers the Saturn IB upper stage is the most powerful hydrogen-fueled engine

to be developed for flight. It represents a state-of-the-art advance including new systems concepts and significant improvements in many component designs.

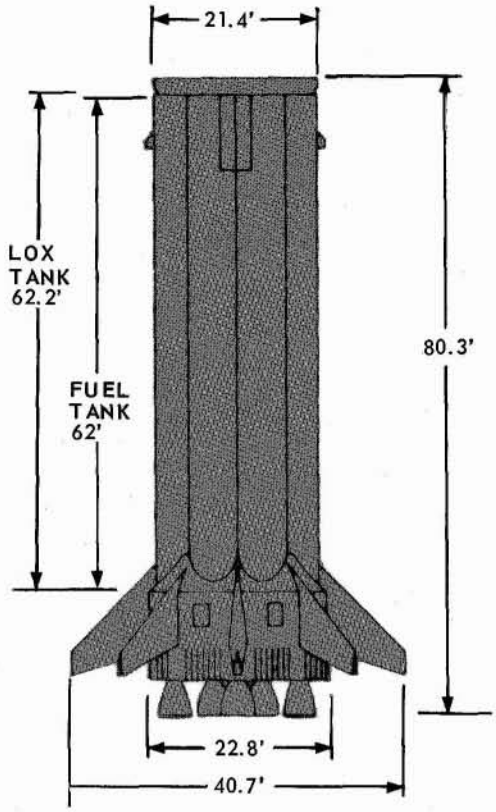
Development of a large engine using liquid hydrogen, with a self-integrated control circuit and system, and a self-contained instrumentation assembly, along with stringent requirements for starting and re-starting at altitude with long coast times between starts, required investigation of new areas of circuitry, high-speed rotating machinery, and unique thrust chamber designs.

Control circuits and valves were developed to assure utilization of propellants at maximum efficiency and to permit changing the ratio of oxidizer to fuel in the engine during operation. These not only make it possible to control propellant depletion, but also to vary the engine's thrust by changing the oxidizer-fuel mixture ratio. As the mixture ratio is changed from a nominal 5.0 to a maximum of 5.5 or a minmum of 4.5, thrust varies from about 175,000 to 225,000 pounds.

A pressurized gas sphere is provided for engine start and re-start. It is recharged during test or flight, to remain ready at the proper pressure for re-start. Electrical controls are sequenced during the initial start and burn, in order to re-set the system for another start. The electrical package contains circuitry to permit the re-starting, and engine conditioning controls have been established to provide proper temperature and pressures of the fluids in the engine at the re-start signal.

Significant advances were made in the design of such components as a regeneratively cooled thrust chamber that permits proper cooling at the minimum and maximum flow; an injector that gives optimum performance through the entire thrust range; an axial flow turbopump to feed liquid hydrogen in high volume, and a centrifugal oxidizer pump that is separately operated; a gas generator to produce gases for operation of the fuel and oxidizer turbopumps; and new types of insulation and advanced circuitry.

S-IB STAGE FACT SHEET



WEIGHT: 84,100* lb (empty)
995,000* lb (loaded)
BURN TIME: 145* sec
VELOCITY: 7700* ft/sec (maximum)
ALTITUDE AT BURNOUT: 39* statute miles

MAJOR STRUCTURAL COMPONENTS

TAIL UNIT ASSEMBLY
NINE PROPELLANT CONTAINERS
SPIDER BEAM UNIT ASSEMBLY
EIGHT FIN ASSEMBLIES

MAJOR SYSTEMS

PROPULSION: Eight bipropellant H-1 engines
Total thrust: 1,600,000 lb (S-IB-4 and S-1B-5); 1,640,000 lb (S-IB-6 through S-IB-12)
Propellant: RP-1 — 281,600 lb (42,100 gal)*
LOX — 629,900 lb (66,900 gal)*
Pressure: Control, 1.0 cubic foot of gaseous nitrogen at 3,000 psig.
Fuel pressurization, 38.6 cubic feet of gaseous helium at 3,000 psig.
LOX pressurization, gaseous oxygen converted from LOX by engines.
HYDRAULIC: Power for gimbaling four outboard engines.
ELECTRICAL: Two 28 vdc batteries, basic power for all electrical functions.
TRACKING: ODOP Transponder.
TELEMETRY: Four subsystems handling 396 flight measurements (S-IB-4).
Two subsystems handling approximately 230 flight measurements (S-IB-5 through S-IB-12).

*Approximations for Stages S-IB-4 through S-IB-12. Refer to Saturn IB Fact Sheet for more exact values.

S-IB STAGE

S-IB STAGE DESCRIPTION

The S-IB stage consists, basically, of a cluster of eight H-1 rocket engines (4 fixed inboard and 4 steerable outboard), a tail unit assembly, nine propellant containers, a spider beam unit assembly, and eight fin assemblies. The nine propellant containers attach to the tail unit assembly at the lower end and to the spider beam unit assembly at the top.

S-IB STAGE FABRICATION AND ASSEMBLY

Construction of the S-IB stage begins at the MSFC Michoud Assembly Facility with the fabrication of the tail unit assembly and the spider beam unit assembly. The tail unit assembly, propellant containers, and spider beam unit assembly are then brought together in a major assembly operation called clustering. After clustering, the eight engines and various pneumatic, mechanical, and electrical systems are installed to complete the assembly of the stage.

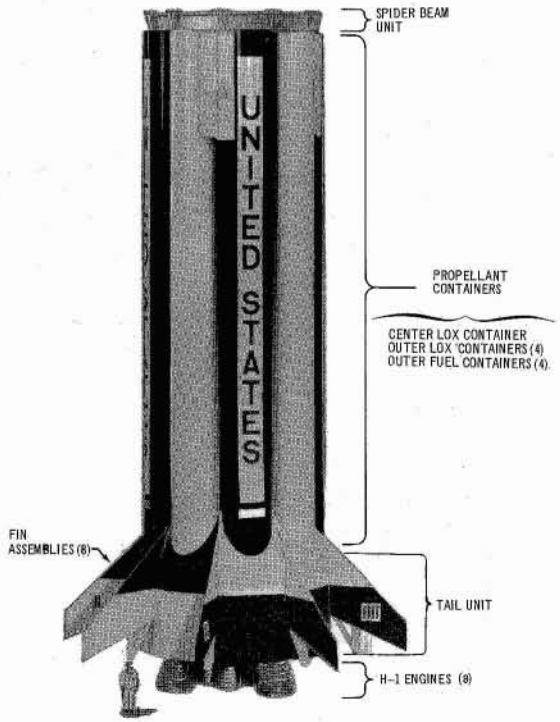
Tail Unit Assembly

The tail unit assembly consists of four radial thrust support outriggers and four radial fin support outriggers, all of which are attached to a barrel assembly core. The ends of the outriggers are joined by shroud panels which form the periphery of the tail unit. A lower shroud panel assembly encloses the engines and forms the engine compartment. The forward end of the assembly is closed by fire walls and the aft end is closed by flame shields. When assembled into the stage, four of the engines are attached to the barrel assembly and four are attached to the thrust outriggers.

The barrel assembly is comprised of an upper and a lower thrust ring, four shear web assemblies, and four skin panels.

The thrust support outrigger assemblies consist of two shear panels, a thrust beam, an actuator support beam, an outboard engine mounting pad, three webs, a bulkhead, a shroud support plate, and various angles and channels. The fin support and thrust support outriggers are similar. The fin support outriggers have no thrust support beam and no actuator support beam.

The water quench system, calorimeter purge system, and fire detection system, along with various lines, components, and electrical equipment that are parts of other stage systems are installed to complete the tail unit assembly.



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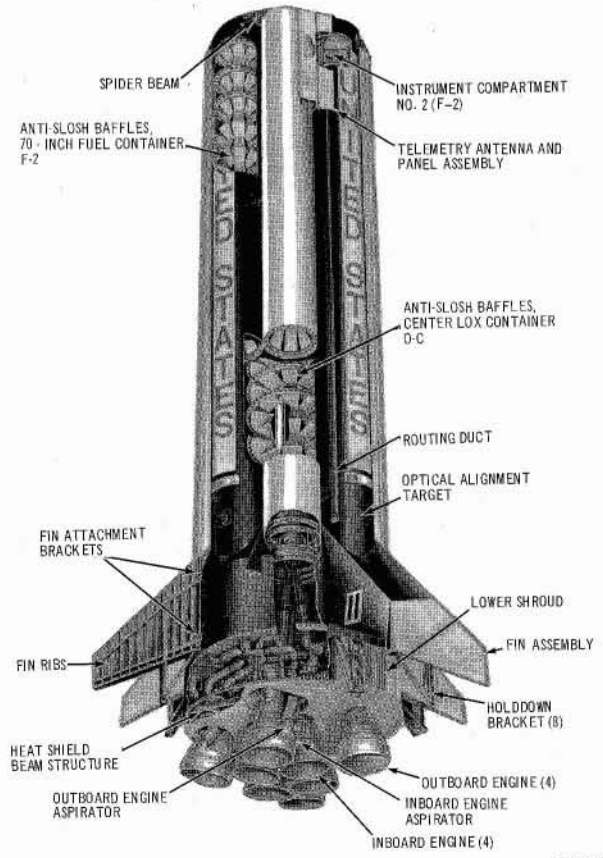
S-IB Stage



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S-IB Stage Assembly Cutaway

LEGEND FOR S-IB STAGE MANUFACTURING SEQUENCE FLOW CHART

1 UPPER AND LOWER THRUST RING
- FABRICATE AND ASSEMBLE

2 ENGINE SHEAR WEB ASSEMBLY
- FABRICATE AND ASSEMBLE

3 BARREL ASSEMBLY
- ASSEMBLE UPPER AND LOWER THRUST RINGS TO SHEAR WEBS
- INSTALL INTERMEDIATE RINGS AND SUPPORT FITTINGS
- INSTALL SKIN ASSEMBLIES AND FIREWALL PANELS
- INSTALL ATTACHING HARDWARE INCLUDING ENGINE MOUNTING PADS

4 FIN SUPPORT OUTRIGGER ASSEMBLY
- ASSEMBLE SHEAR PANELS, DRILL FIN AND BARREL CONNECTING HOLES
- POSITION AND INSTALL WEBS
- INSTALL ATTACHING HARDWARE, STIFFENERS, AND FITTINGS
- INSTALL SHROUD SUPPORT PLATES

5 THRUST SUPPORT OUTRIGGER ASSEMBLY
- ASSEMBLE SHEAR PANELS TO THRUST BEAM
- INSTALL BULKHEAD
- INSTALL STIFFENERS, DRILL FIN AND BARREL CONNECTING HOLES
- INSTALL FITTINGS
- INSTALL SHROUD SUPPORT PLATE
- INSTALL ACTUATOR SUPPORT BEAM

6 TAIL SECTION ASSEMBLY
- INSTALL, ALIGN, AND ASSEMBLE OUTRIGGERS TO BARREL ASSEMBLY } THRUST STRUCTURE ASSEMBLY
- INSTALL RING SEGMENTS, BRACKETRY, AND FITTINGS } THRUST STRUCTURE ASSEMBLY
- INSTALL FUEL AND LOX BAY FIREWALL PANELS } THRUST STRUCTURE ASSEMBLY
- INSTALL UPPER SHROUD PANELS
- INSTALL LOWER SHROUD PANELS } LOWER SHROUD ASSEMBLY
- INSTALL HEAT SHIELD BEAM STRUCTURE AND HEAT SHIELD PANELS } LOWER SHROUD ASSEMBLY
- INSTALL TOOLING RING

7 TAIL UNIT ASSEMBLY
- INSTALL WATER QUENCH SYSTEM
- INSTALL LOX PRESSURIZATION SYSTEM LINES
- INSTALL ENGINE PURGE LINES
- POSITION FUEL AND LOX SUCTION LINES
- INSTALL CALORIMETER PURGE SYSTEM
- INSTALL LOX REPLENISH LINES
- INSTALL STATIC TEST NITROGEN PURGE LINES
- INSTALL FIRE DETECTION SYSTEM
- INSTALL ELECTRICAL EQUIPMENT
- INSTALL MISCELLANEOUS EQUIPMENT
- INSPECT AND PERFORM CONTINUITY, MEGGER, AND PNEUMATIC CHECKOUT

8 105-INCH LOX CONTAINER PRIOR TO CLUSTERING
- INSTALL MISCELLANEOUS EQUIPMENT
- INSTALL EXTERNAL MECHANICAL EQUIPMENT
- INSTALL SNOW AND ICE SHIELDS

9 SPIDER BEAM UNIT ASSEMBLY
- ATTACH UPPER AND LOWER SPLICE PLATES TO HUB ASSEMBLY: ATTACH RADIAL BEAMS, CROSS BEAMS, AND INSTALL SPLICE PLATES
- DRILL HOLES FOR PROPELLANT CONTAINER CLUSTERING AND S-IVB STAGE ADAPTATION
- PREFIT AND REMOVE SEAL PLATES
- INSTALL FORWARD TOOLING RING
- INSTALL OPTICAL TARGETS AND ALIGN
- INSTALL CONTROL AND MEASURING COMPONENTS, FITTINGS, AND TUBING

10 70-INCH LOX CONTAINERS PRIOR TO CLUSTERING
- INSTALL MISCELLANEOUS HARDWARE
- INSTALL EXTERNAL MECHANICAL EQUIPMENT
- INSTALL ENVIRONMENTAL PROTECTION EQUIPMENT
- INSTALL BREATHER ASSEMBLIES
- PERFORM CONTINUITY, MEGGER, AND PNEUMATIC CHECKOUT

11 70-INCH FUEL CONTAINERS PRIOR TO CLUSTERING
- INSTALL MISCELLANEOUS HARDWARE
- INSTALL ELECTRICAL COMPONENTS IN CONTAINER SKIRTS AND INSTRUMENT COMPARTMENTS OF FUEL TANKS F-1 AND F-2
- INSTALL EXTERNAL MECHANICAL EQUIPMENT
- INSTALL ELECTRICAL TRUNK CABLES AND CONDUCT COVER ASSEMBLIES
- INSTALL HIGH PRESSURE SPHERE IN FORWARD SKIRT OF F-3 AND F-4
- INSTALL BREATHER ASSEMBLIES
- PERFORM CONTINUITY, MEGGER, AND PNEUMATIC CHECKOUT

12 CLUSTERING
- POSITION TAIL UNIT IN ASSEMBLY FIXTURE
- ATTACH 105-INCH LOX CONTAINER TO TAIL UNIT ASSEMBLY
- ATTACH SPIDER BEAM TO 105-INCH LOX CONTAINER
- INSTALL LOX CONTAINERS IN OPPOSITE PAIR SEQUENCE
- INSTALL FUEL CONTAINERS IN OPPOSITE PAIR SEQUENCE
- INSTALL INTERCONNECTING TUBING
- INSTALL 60-DEGREE FAIRINGS
- PERFORM ALIGNMENT CHECKS

13 TAIL AREA INSTALLATION – PHASE I
- INSTALL LOX AND FUEL PREVALVES
- INSTALL LOX SUMP INTERCONNECT LINES
- INSTALL FUEL SUMP INTERCONNECT LINES
- INSTALL PNEUMATIC TUBING
- INSTALL LOX REPLENISHING VALVE
- INSTALL DELUGE PURGE AND WATER QUENCH SYSTEM TUBING

14 TANK TOP AREA INSTALLATION – PHASE I
- INSTALL LOX PRESSURIZATION SYSTEM LINES AND VALVES
- INSTALL FUEL PRESSURIZATION SYSTEM LINES AND VALVES
- INSTALL COOLING DUCTS FOR INSTRUMENT COMPARTMENT COOLING SYSTEM
- INSTALL FLIGHT MEASUREMENT TUBING

15 ENGINE MODIFICATION
- REWORK ENGINE PER INSPECTION REPORT
- MODIFY ENGINE PER MODIFICATION DRAWING
- INSTALL HYDRAULIC SYSTEM (OUTBOARD ENGINES ONLY)
- INSTALL FLIGHT MEASUREMENT SYSTEM
- WEIGH AND DETERMINE CENTER OF GRAVITY

15A INBOARD ENGINE INSTALLATION
- INSTALL INBOARD ENGINES NO. 5, 6, 7, AND 8, FLAME SHIELD, AND ACCESS CHUTE
- INSTALL INBOARD ENGINE FUEL AND LOX SUCTION LINES

16 TAIL AREA INSTALLATION – PHASE II
- INSTALL FLIGHT ELECTRICAL EQUIPMENT
- INSTALL INBOARD ENGINE TURBINE EXHAUST
- INSTALL INBOARD ENGINE DRAIN LINES
- INSTALL INBOARD ENGINE HEAT EXCHANGER LOX AND GOX LINES
- CONNECT INBOARD ENGINES TO PURGE SYSTEM
- INSTALL ELECTRICAL CABLES

17 OUTBOARD ENGINE INSTALLATION
- INSTALL OUTBOARD ENGINES NO. 1, 2, 3, AND 4
- INSTALL OUTBOARD ENGINE FUEL AND LOX SUCTION LINES

18 TANK TOP AREA INSTALLATION – PHASE II
- INSTALL ANTENNA PANELS, ANTENNAS, AND COAXIAL CABLES
- INSTALL STATIC FIRE TUBING
- INSTALL FLIGHT ELECTRICAL EQUIPMENT
- INSTALL STRAIN GAGES
- INSTALL ELECTRICAL CABLES AND CLEAN TOP TANK AREA

19 TAIL AREA INSTALLATION – PHASE III
- INSTALL STATIC TEST FIRE MEASURING SYSTEM
- INSTALL OUTBOARD ENGINE HEAT EXCHANGER LOX AND GOX LINES
- CONNECT OUTBOARD ENGINES TO ENGINE PURGE SYSTEM
- INSTALL ELECTRICAL CABLES TO FIRE DETECTION SYSTEM
- INSTALL MEASURING RACK (ELECTRONIC) MODULES
- INSTALL INBOARD ENGINE FLAME CURTAINS AND ADJOINING HEAT SHIELD PANELS
- INSPECT SPIDER BEAM, LOX AND FUEL BAY AREAS, ENGINES, PROPELLANT CONTAINERS, AND INSTRUMENT COMPARTMENTS
- INSTALL AND ALIGN ACCELEROMETERS, GYROS AND ERECTION TARGETS
- PERFORM ALIGNMENT CHECK OF STAGE

19A STAGE CHECKOUT
- CONTINUITY TESTS
- POWER DISTRIBUTION TESTS
- MECHANICAL ACCEPTANCE TESTS OF STAGE SYSTEMS
- INSTRUMENTATION AND TELEMETRY TESTS
- MECHANICAL COMPONENTS AND ELECTRICAL NETWORKS TESTS
- SIMULATED PLUG DROP AND ALL SYSTEMS TESTS

20 PREPARATION FOR SHIPMENT AND STATIC TEST
- CLEAN AND PAINT TAIL UNIT
- INSTALL OUTBOARD ENGINE FLAME CURTAINS, AND STATIC TEST HEAT SHIELD PANELS
- DETERMINE PRELIMINARY WEIGHT
- INSTALL RADIATION SHIELD, STATIC TEST ENVIRONMENTAL PROTECTION, ERECTION HARDWARE, SHIPPING INSTRUMENTATION, AND ROAD SHIPMENT PROTECTION EQUIPMENT
- SHIP TO STATIC TEST SITE

21 POST STATIC TEST REFURBISHMENT
- RECEIVE AND CLEAN
- REMOVE STATIC TEST COMPONENTS AND INSTRUMENTATION
- INSTALL FLIGHT COMPONENTS AND INSTRUMENTATION
- PERFORM STAGE ALIGNMENT CHECK
- PERFORM FUNCTIONAL CHECKOUT

21A STAGE CHECKOUT
- CONTINUITY TESTS
- POWER DISTRIBUTION TESTS
- MECHANICAL ACCEPTANCE TESTS OF STAGE SYSTEMS
- INSTRUMENTATION AND TELEMETRY TESTS
- MECHANICAL COMPONENTS AND ELECTRICAL NETWORKS TESTS
- UMBILICAL DISCONNECT, SIMULATED LAUNCH AND FLIGHT ALL SYSTEMS TEST

22 PREPARATION FOR SHIPMENT TO LAUNCH SITE
- INSTALL ADDITIONAL FLIGHT ITEMS
- DETERMINE WEIGHT AND CENTER OF GRAVITY
- CLEAN AND TOUCH-UP EXTERNAL SURFACE
- INSPECT AND REPAIR
- INSTALL ENVIRONMENTAL PROTECTION, SHIPPING INSTRUMENTATION, ERECTION HARDWARE, AND ROAD SHIPMENT PROTECTION EQUIPMENT
- INSPECTION AND FINAL BUY-OFF
- SHIP TO LAUNCH SITE
- SHIP LOOSE ITEMS

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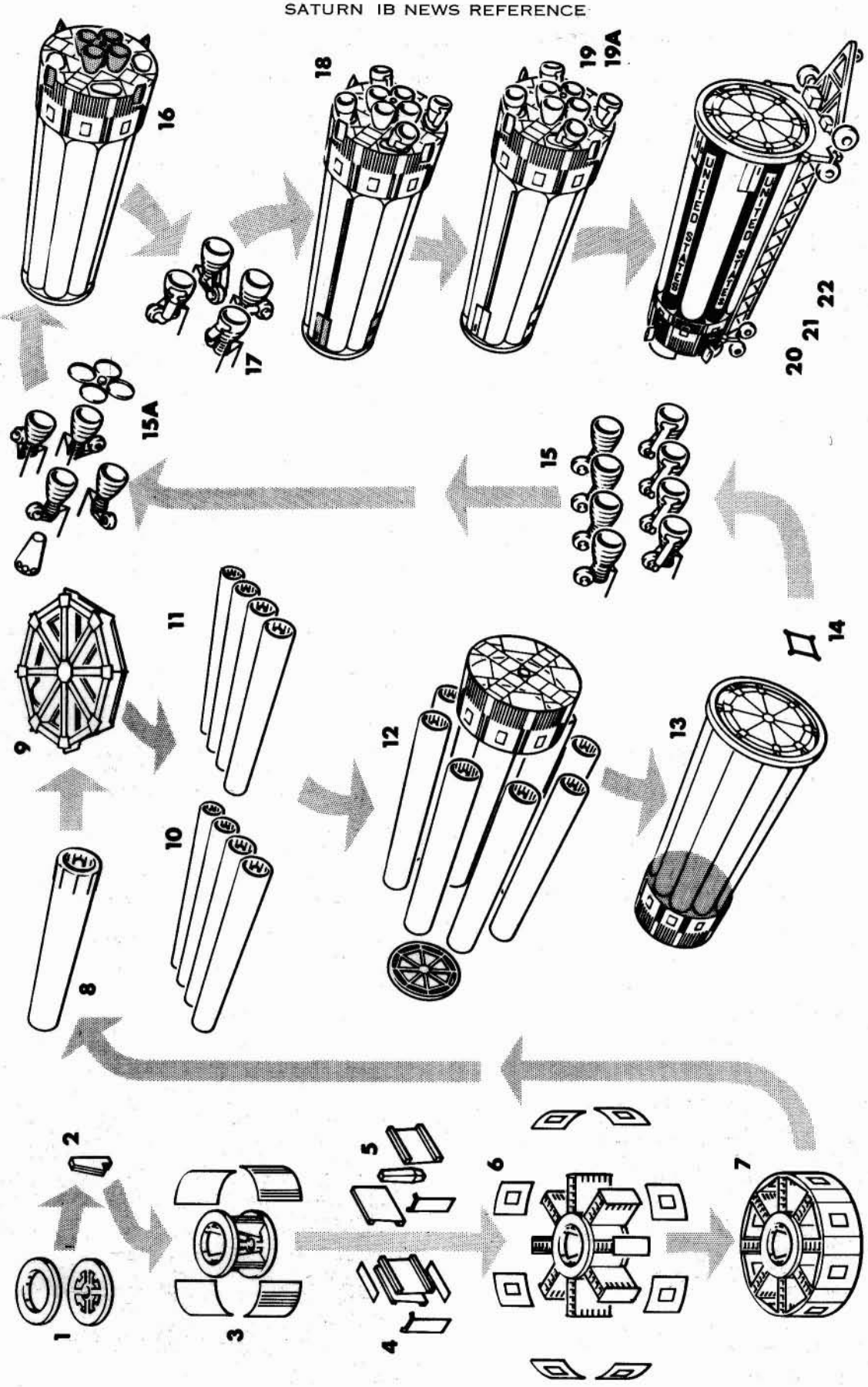
Legend for S-IB Stage Manufacturing Sequence Flow Chart



S-IB Stage Manufacturing Sequence Flow Chart

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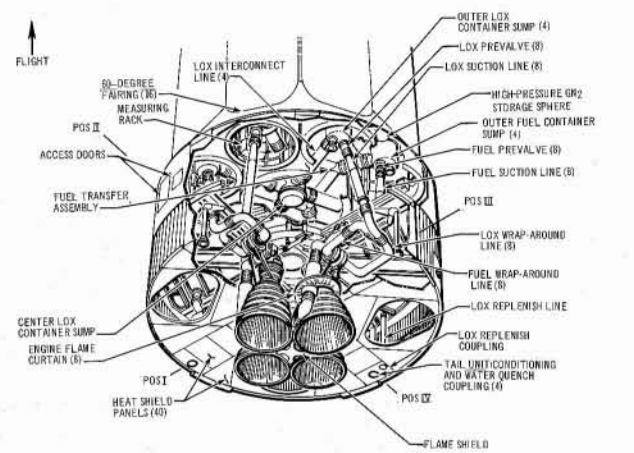
Propellant Containers

The cylindrical section of each propellant container is built up of skin-milled, butt-welded, aluminum alloy segments internally reinforced with rings to form a monocoque type construction. Container wall thickness varies from top to bottom in relation to stress concentrations. Hemispherical bulkheads are welded to each end of the cylindrical section, and a sump is welded to the aft bulkhead. The forward bulkhead of all five LOX containers are fitted with a pressurization and vent manifold; the four fuel containers have openings for fuel or LOX vent manifold connections. A cylindrical skirt reinforced with longerons is attached to both the forward and aft bulkheads to complete the basic container. The units are cleaned, painted, pressure tested, and calibrated for precise volume before shipment to the Michoud Assembly Facility.

Internal and external equipment is installed to modify the basic container. Electrical equipment is installed in the aft skirt areas of all four fuel containers, and in the instrument compartments located in the forward skirts of fuel containers F-1 and F-2. A 19.28-cubic foot high-pressure sphere is installed into the forward skirt of fuel containers F-3 and F-4.

Spider Beam Unit Assembly

The spider beam unit assembly is assembled in a special fixture. A hub assembly is placed in the center of the fixture and the upper and lower splice plates are attached to the hub. Eight radial beams are attached to the hub assembly at 45-degree intervals and the outer ends of the radial beams are joined by cross-beams fastened with splice plates. When the basic structure is completed, hardware is installed that will be used to attach the propellant containers during the clustering operation.



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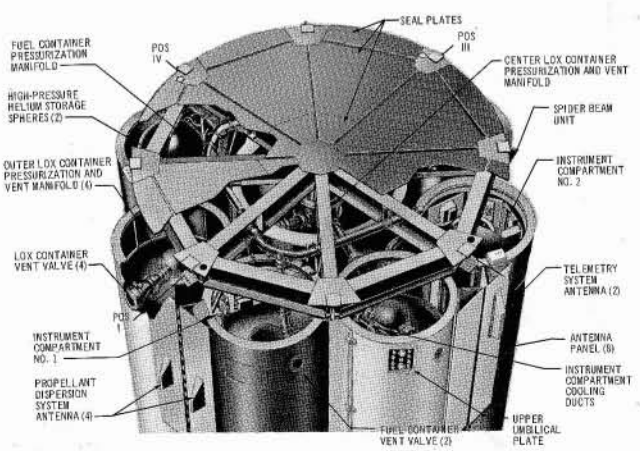
S-IB Stage Tail Area Cutaway


SEAL PLATES
FUEL CONTAINER PRESSURIZATION MANIFOLD
POS IV
POS III
CENTER LOX CONTAINER PRESSURIZATION AND VENT MANIFOLD
HIGH-PRESSURE HELIUM STORAGE SPHERES (2)
SPIDER BEAM UNIT
OUTER LOX CONTAINER PRESSURIZATION AND VENT MANIFOLD (4)
INSTRUMENT COMPARTMENT NO. 2
LOX CONTAINER VENT VALVE (4)
TELEMETRY SYSTEM ANTENNA (2)
POS I
INSTRUMENT COMPARTMENT NO. 1
ANTENNA PANEL (8)
INSTRUMENT COMPARTMENT COOLING DUCTS
PROPELLANT DISPERSION SYSTEM ANTENNA (4)
FUEL CONTAINER VENT VALVE (2)
UPPER UMBILICAL PLATE

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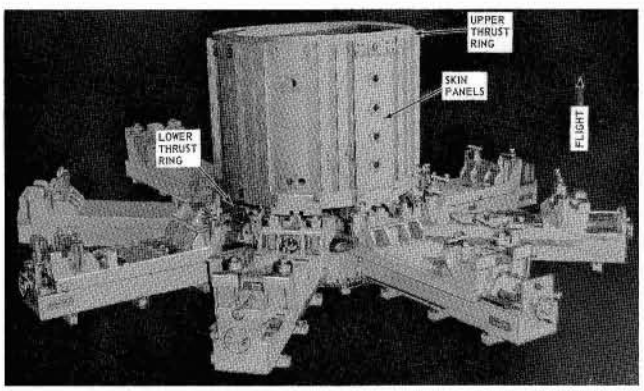
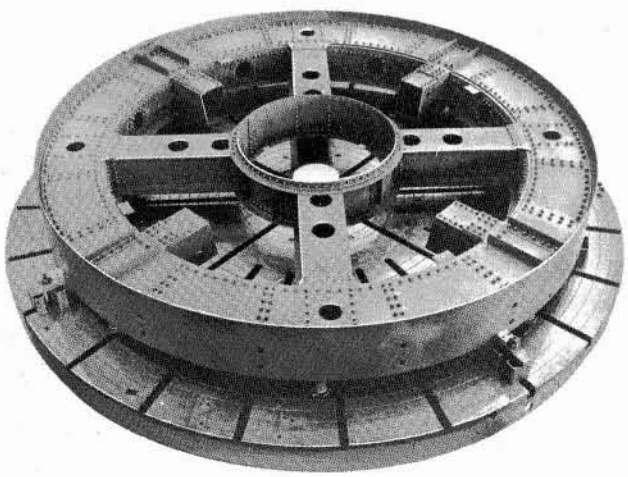
S-IB Stage Top View Cutaway



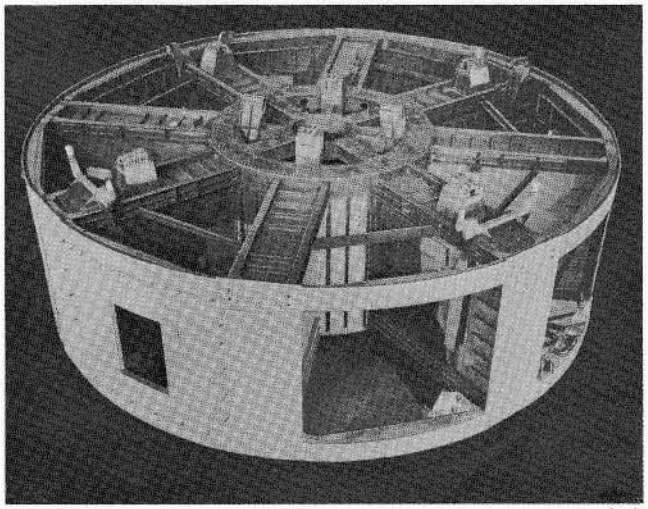
CC-04(D)

Barrel Assembly — The barrel assembly is the central structural member of the thrust structure assembly. The barrel assembly is shown mounted on an assembly fixture.



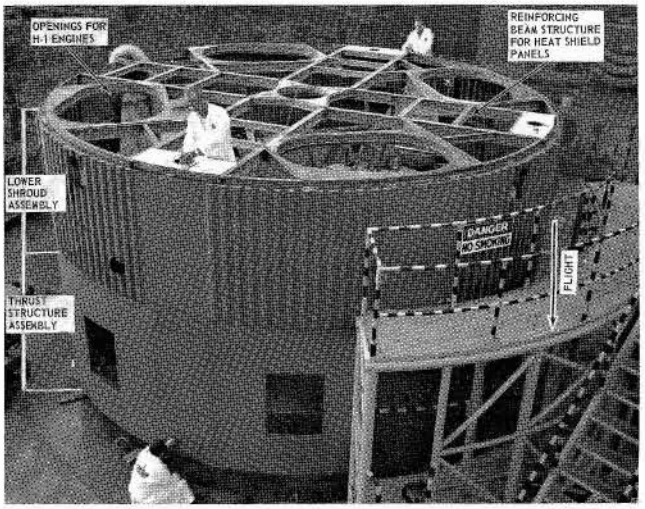
CC-04(B)

Lower Thrust Ring on Alignment Table — Lower thrust ring, completely assembled, is shown on an alignment table ready for optical alignment checkout prior to fabrication of the barrel assembly.



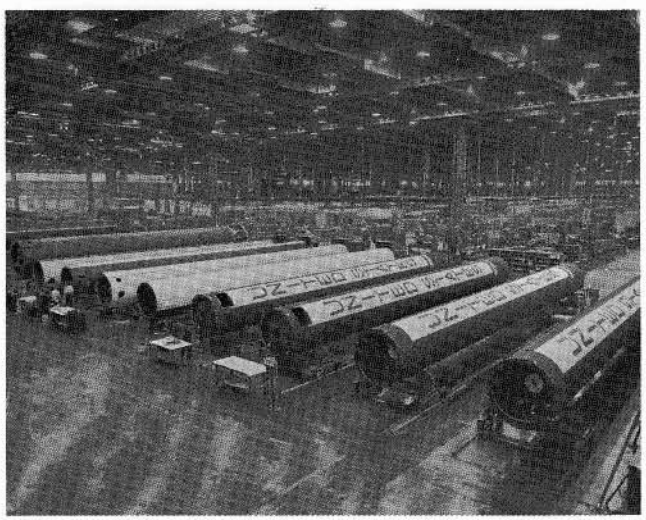
CC-03 (C)

Thrust Structure Assembly — The thrust structure assembly is the mounting point for the engine, the propellant containers, and the fins. Engine thrust is transmitted through the thrust structure assembly.



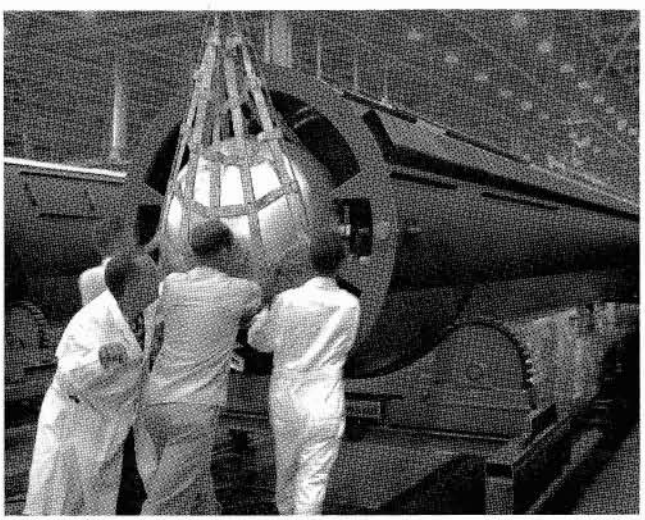
CC-02 (B)

Tail Unit Assembly — Chrysler technicians inspect the tail unit assembly during an intermediate phase of fabrication. The tail unit is inverted on an assembly fixture in this photograph.



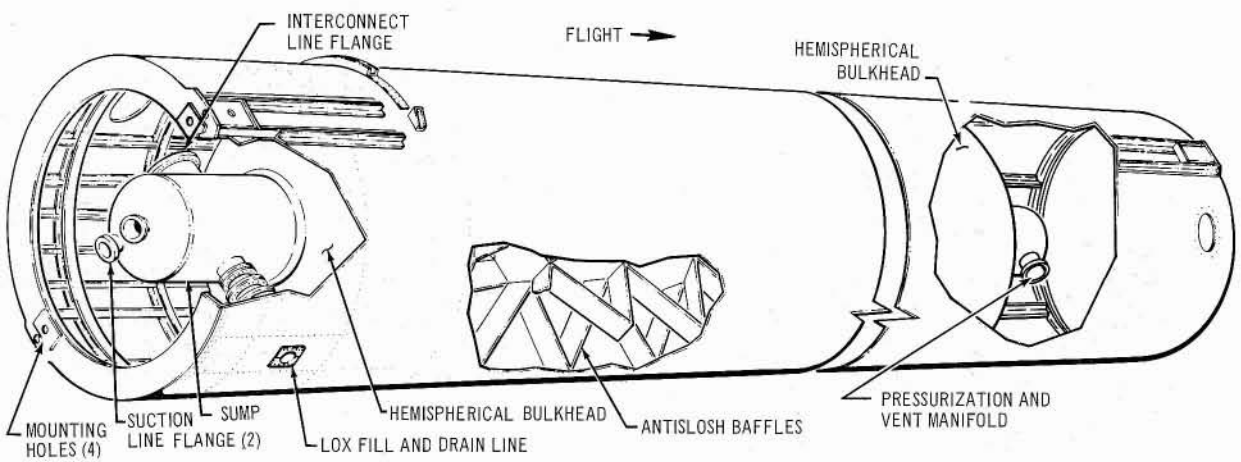
CC-06 (D)

Propellant Containers — All nine propellant containers are shown prior to clustering. From left to right are the 105-inch diameter center LOX container, four 70-inch diameter outer LOX containers, and four 70-inch diameter outer fuel containers.



CC-06 (F)

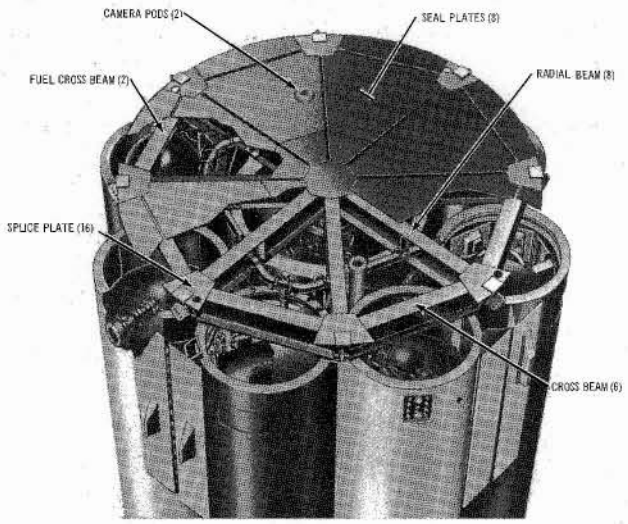
Installing Fuel Container Pressurization Sphere — Workmen install a pressurization sphere into the forward skirt of a fuel container. The sphere is used to store helium for the fuel pressurization system. Two fuel containers are equipped with spheres and two have instrument compartments in the forward skirts.



CC-06 (E)

Outer Propellant Container — This cutaway drawing of LOX container 0-3 is representative of all 70-inch propellant containers; however, only containers 0-3 and F-1 are equipped with a fill and drain line. The four fuel containers do not have the pressurization and vent manifold.

Holes are drilled for use in attaching the S-IVB stage to the spider beam at the launch site. A special handling ring is attached to the spider beam and an alignment check is made in preparation for the clustering operation.



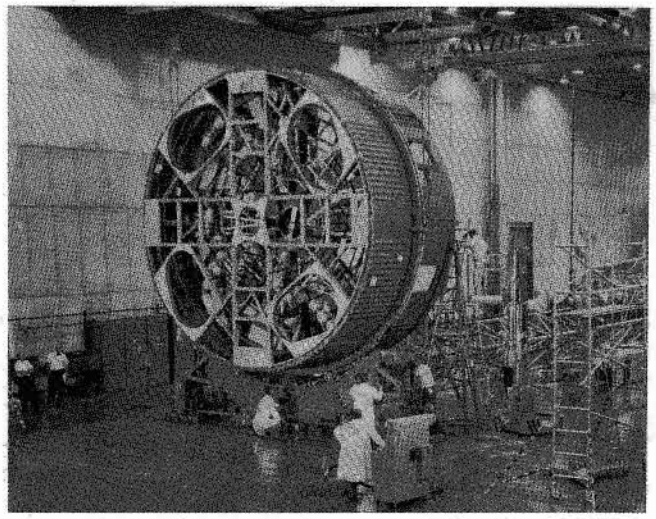
CC-08 (B)

Forward End of S-IB Stage — The forward end of the S-IB stage is held together by the spider beam unit assembly. The forward skirts of the LOX containers are rigidly fastened to the spider beam to give the forward end of the stage structural integrity. The fuel tanks are mounted on a sliding pin arrangement that allows them to compensate for LOX tank contraction.

Clustering

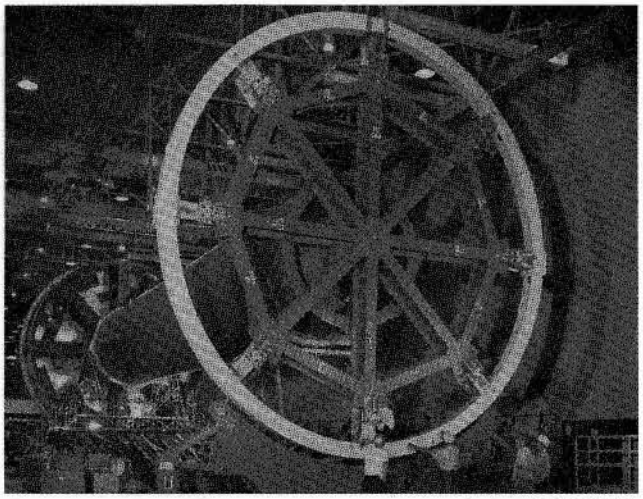
The S-IB stage is clustered in a horizontal position. Multiple-level platforms at the ends of the stage allow simultaneous assembly operations.

The assembly fixture for the S-IB stage is a truss structure with a front and a rear cradle which is supported by four adjustable leveling stands that allow vertical and horizontal movement for the



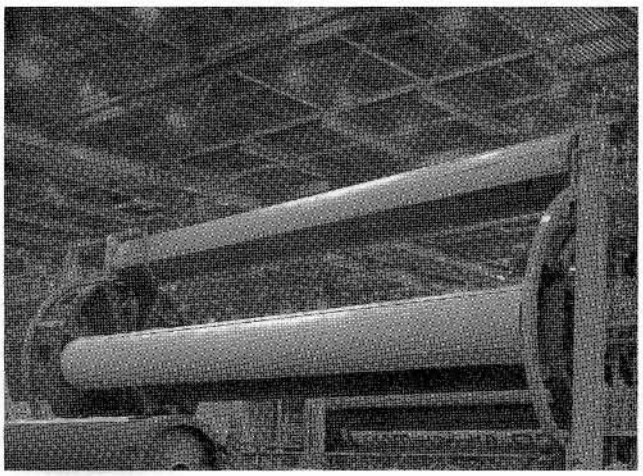
CC-09 (A)

Positioning Tail Unit Assembly — The tail unit is the first major assembly to be placed in the assembly fixture.



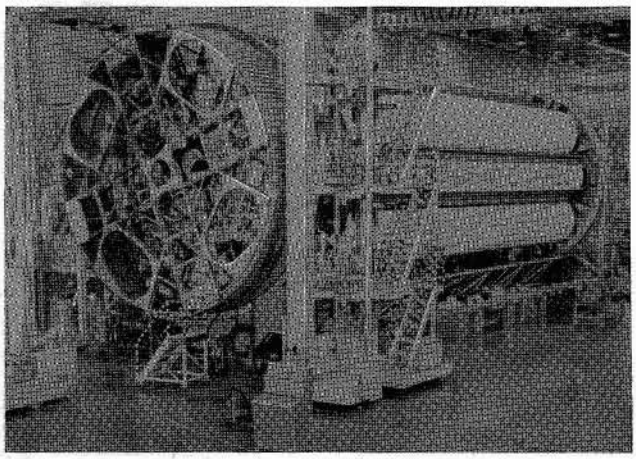
CC-09 (D)

Positioning Spider Beam Unit Assembly — Workmen position the spider beam on the assembly fixture. The man wearing earphones is directing an overhead crane that maneuvers the assembly.



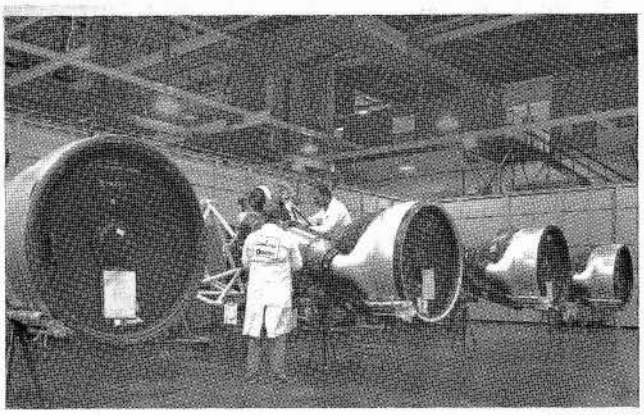
CC-09 (E)

Container Clustering — The third of four LOX containers is lowered into position using two overhead cranes. The LOX containers are clustered first, followed by the fuel containers. The containers are clustered in an opposite pair sequence to keep the assembly balanced.



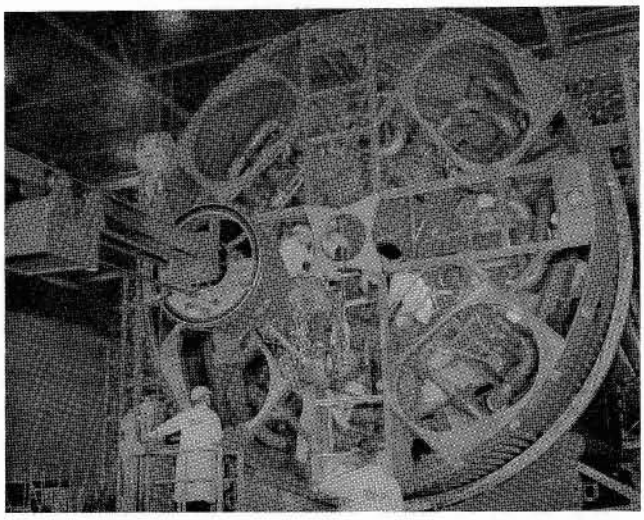
CC-09 (C)

Clustering Complete — The stage is shown completely clustered in this illustration. The next major operation is the installation of the H-1 engines. The fin assemblies are not permanently installed until the stage is erected on the launch pad. The console in the foreground is used to raise, lower, and rotate the stage in the assembly fixture.



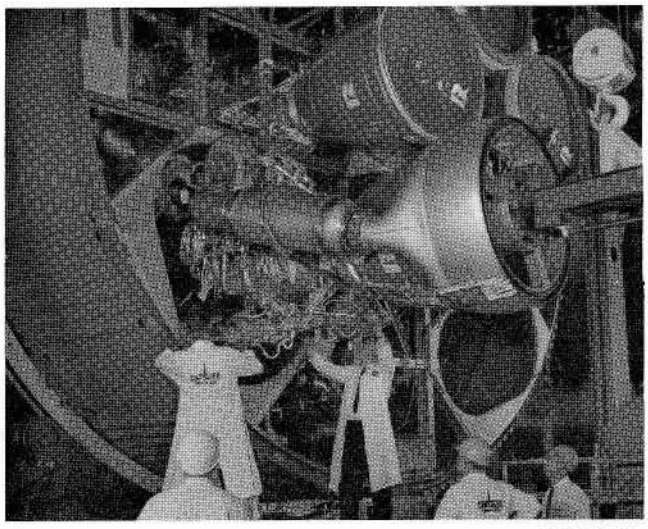
CC-10(A)

Engine Modification — The basic H-1C (inboard) and H-1D (outboard) engines, produced by Rocketdyne, undergo modification by Chrysler before they are installed on the stage. In this illustration one of four outboard engines is undergoing modification.



CC-05(B)

Inboard Engine Installation — Chrysler technicians, using a special handling tool, install the first of four inboard H-1 engines. The large diameter lines in the background with 90-degree bends are the fuel and LOX suction lines through which propellant flows to the engines.



CC-05(C)

Outboard Engine Installation — Technicians install the first of four outboard engines into the tail unit assembly.

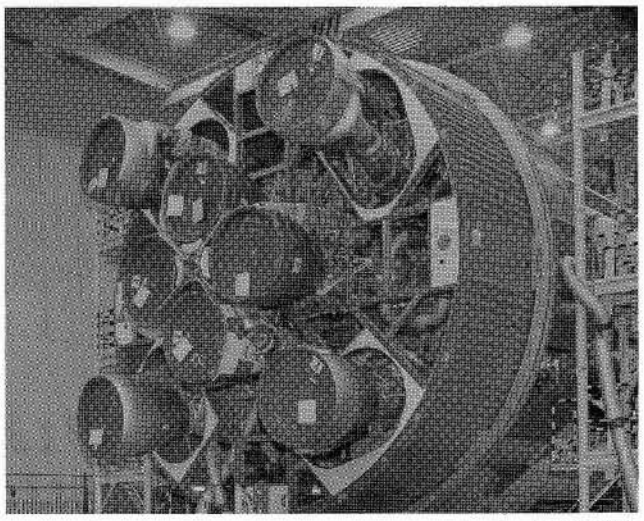
alignment of components. Rollers in the front and rear cradles allow the stage to be rotated during assembly.

A large bridge crane is used to place the tail unit assembly in the rear cradle. The 105-inch diameter center LOX container is positioned so that the aft skirt can be fastened to the top end of the barrel assembly. The spider beam unit assembly is then attached to the forward end of the center LOX container. The tail unit assembly, center LOX container, and spider beam unit assembly are rotated as a unit to facilitate attachment of the outer containers.

The 70-inch diameter outer LOX containers are installed first. The containers are installed in an opposite-pair sequence to keep the assembly within the balance requirements of the assembly fixture. LOX containers are attached to the tail unit assembly and to the spider beam unit assembly with retaining bolts and eyebolt assemblies.

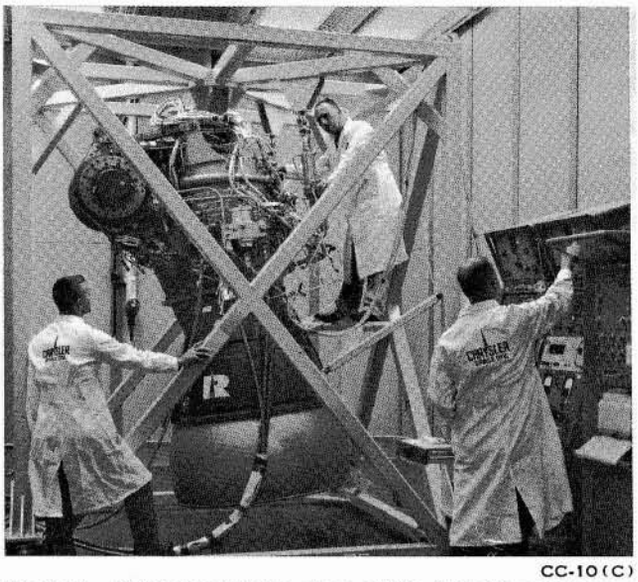
The fuel containers are installed in a similar manner to complete the container clustering operation. Fuel containers are attached to the tail unit assembly by a ball and socket arrangement, and to the spider beam unit assembly by a sliding pin and socket arrangement. This allows for the variable distance between the spider beam unit assembly and the thrust structure assembly caused by contraction of the LOX tanks at cryogenic temperatures.

Assembly operations are completed by installing lines, manifolds, cables, electrical and pneumatic components, engines, and instruments. The fin assemblies are attached to the stage as the vehicle is erected at the launch site.



CC-05(D)

H-1 Engines — The eight H-1 engines are clustered in an inboard and an outboard square pattern. The outboard square pattern is rotated 45 degrees from the inboard pattern.



CC-10(C)

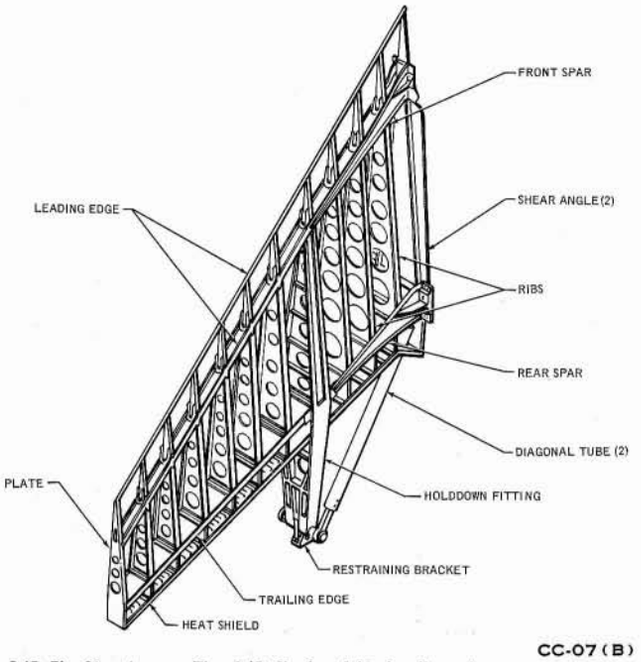
Checkout — Chrysler technicians check out the hydraulic system on an outboard engine before it is installed on the stage.

H-1 ENGINE INSTALLATION

Before being installed in the stage, the H-1 engines undergo modification, consisting primarily of the installation of static test and flight measurement instrumentation and cables on inboard and outboard engines, and the installation of the hydraulic system on outboard engines.

The inboard engines are installed first, using a special handling fixture to insert each engine into the tail unit assembly. Fuel and LOX suction lines are installed and, when the last inboard engine is installed, the flame shield is installed and a series of operations connecting the inboard engines into other stage systems is performed. These operations include the installation of engine drain lines, GOX lines, purge systems, LOX supply lines to heat exchangers, and electrical cables.

The outboard engines are installed in approximately the same manner as the inboard engines. The engines are connected into the stage systems, and the electrical cables that monitor and control all eight engines are installed. Engine installation is considered complete after the flexible flame curtains, the heat shield panels, and other heat protective equipment has been installed.



CC-07(B)

S-IB Fin Structure — The S-IB fin is of basic rib and spar construction, covered with aluminum skin panels. The eight fins support the launch vehicle on the launch pad and provide aerodynamic stability during flight.

CC-05(E)

Heat Shield Panel Installation — Chrysler technicians install heat shield panels on the lower shroud assembly beam structure. The panels enclose the aft end of the tail assembly to form an engine compartment. An outboard engine flame curtain is shown in the right foreground.

CC-11(C)

Installation — Technician laces electrical harness located in the aft skirt of a propellant container.

FIN ASSEMBLIES

Each of eight fin assemblies is built around a front and rear spar, a hold down fitting, and two diagonal tubes which strengthen the structure. A leading edge is attached to the front spar and a trailing edge to the rear spar. A heat shield is attached to the trailing edge to protect the fin from engine exhaust. Skin panels are riveted to the fin structure to form a smooth aerodynamic surface.

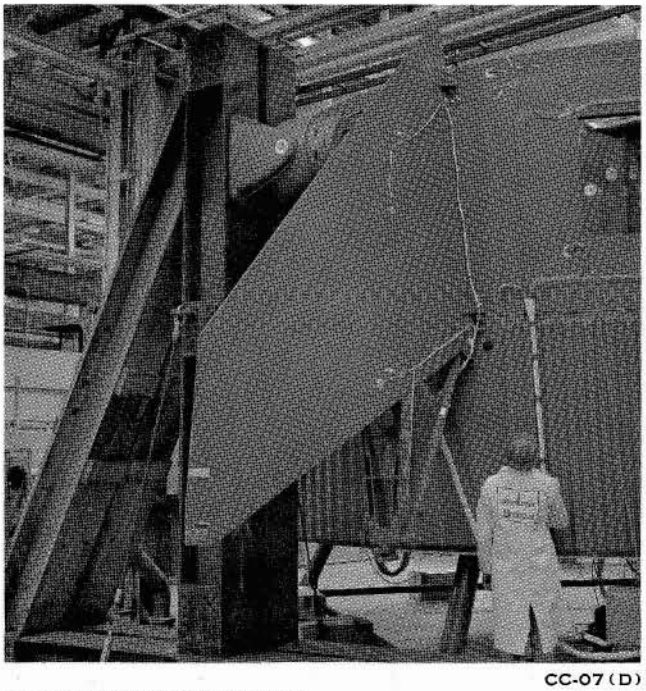
The basic fin assembly is completed by installing instrumentation such as calorimeters, temperature gages, and strain gages which are part of the measuring program.

Electrical Fabrication and Assembly

The S-IB booster stage is much more than eight rocket engines and a cluster of propellant containers. In addition to thousands of feet of tubing and numerous mechanical and electromechanical valves and regulators, the S-IB booster stage contains 53 miles of wire terminating in approximately 73,000 electrical connections, that tie together almost 1,700 electrical and electronic components.

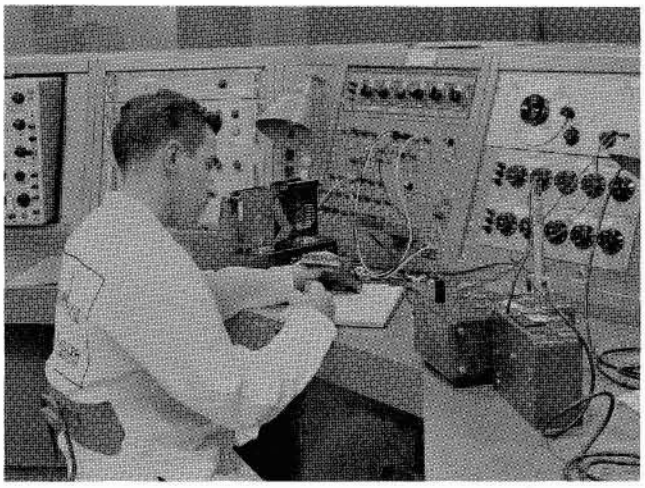
All of the electrical cables are fabricated by Chrysler. Automatic magnetic-tape programmed instruments check the assemblies for continuity between pins and perform high-voltage leakage tests.

Components of some of the major electronic units, such as a telemetry assembly, a multiplexer, or a timer assembly, are purchased from vendors. The components are assembled by Chrysler, qualified as a unit, functionally tested, and installed on the stage. Other units such as measuring racks, liquid level adapter racks, and distributors are fabricated by Chrysler.



CC-07 (D)

S-IB Fin Undergoing Vibration Test



CC-11 (E)

Test — Electronic assemblies undergo test and calibration in a Chrysler laboratory.

S-IB Stage Checkout

Chrysler technicians and engineers conduct tests on the propulsion, electrical, control, instrumentation, and telemetry systems of the S-IB stage in accordance with the final acceptance test procedures. The tests are performed just prior to shipment of the stage to MSFC, Huntsville for static fire and again just prior to shipment of the stage to KSC, Cocoa Beach, Florida, for launch.

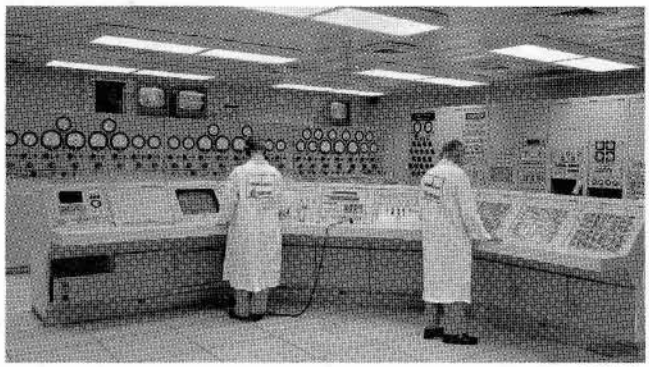
Dual checkout stations located adjacent to the final assembly area permit two stages to be checked out simultaneously. Each checkout station consists of a checkout bay and a control room.

A central computer complex and two telemeter ground stations support the operation of the checkout stations.

The components and systems within each stage are subjected to a series of tests to demonstrate the acceptability of each system and family of systems. A summary of these tests is presented in the following tabulation. The order of presentation is not necessarily the order in which the tests are made since many tests are run simultaneously.

Antennas — Antennas, power dividers, and transmission lines are checked for insulation resistance, continuity, attenuation, and voltage standing wave ratio.

Power Distribution — Resistance checks of the checkout station equipment and stage busses are performed before power is applied. Buss electrical isolation and distribution is verified with the application of power to each buss.



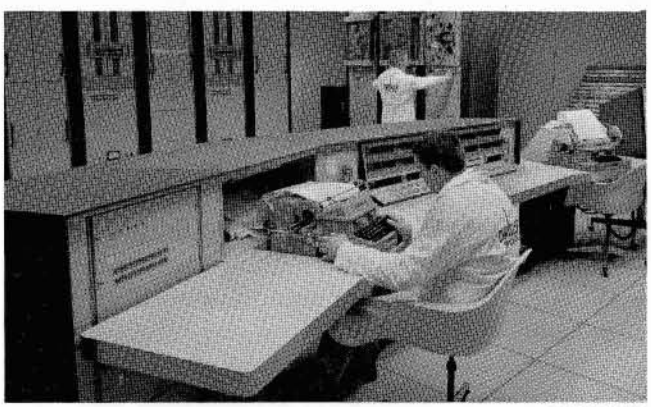
CC-17 (B)

Checkout Station Control Room — The test conductor monitors the progress of activity in the checkout bay on the closed circuit TV monitor. A technician assures that the proper sequence of events occurs by monitoring the panel at the right end of the console.

Instrumentation — Two categories of instrumentation are made.

1. Indirect measurements utilizing signal conditioners for evaluation of acoustic, temperature, pressure, vibration, liquid level, current, strain, and turbine rpm measurements.
2. Direct measurements using a direct output in the flight mode to evaluate acceleration, pressure, and signal levels.

Mechanical Systems — Fuel, oxidizer, propulsion, pneumatic, and hydraulic systems are pressure tested to verify system integrity and to assure that leakage is within the allowable specifications. While these systems are pressurized, other system characteristics such as pressure switch actuation and de-actuation levels, valve timing characteristics, vent valve cracking and reseating pressures, and critical orifice flow rate are checked.

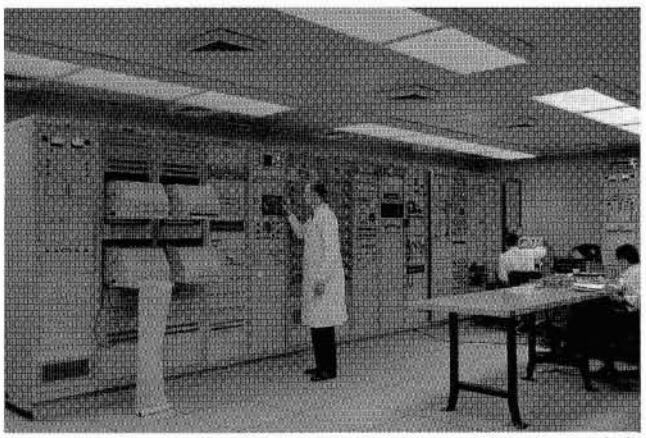


CC-17 (E)

Central Computer Complex — The central computer complex is used to automatically sequence and monitor vehicle checkout. One master and four slave computers provide the required control and monitoring functions. The master control console is located to the right of the operator in the foreground. This operator is seated at one of the two Flexowriters in the computer complex. The operator in the background is operating the magnetic tape recorder. The master computer is programmed through either Flexowriter on the magnetic tape. In turn, the master computer controls the slave computers to sequence and monitor vehicle tests.

Electrical Networks — Station equipment and stage redundant cutoff circuits are checked for the capability to initiate engine cutoff at any time a malfunction is detected. The flight sequence operation is checked by verifying circuit operations that occur during prelaunch, liftoff, and separation functions. Engine cutoff is also verified by simulating the conditions of an engine failure, failure to launch due to loss of liftoff circuitry, and a malfunction after liftoff causing emergency engine cutoff.

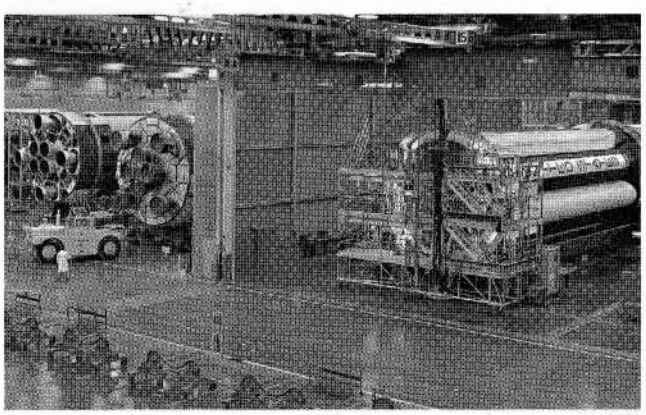
RF Systems — Tests are made to assure proper operation of the telemetry systems and the ODOP (offset doppler) tracking system. The tests assure that power output, voltage standing wave ratio, frequency, insertion loss and doppler shift are within specifications.



CC-17 (F)

Telemeter Ground Test Station — This illustration shows one of the two TM ground stations that support the checkout station. The technician in the center is shown making an output selection on patch board that will be monitored on the oscilloscope located just below. Four oscillograph recorders are shown to the right of the technician.

Radiated Compatibility and Interference — Tests are performed to verify that radiated frequencies of the telemetry system do not interfere with the operation of the command system.



CC-63

S-IB Stage Final Assembly Area

Control System — The outboard engines are tested to assure proper function of hydraulic actuators, control system linearity, and proper reaction to control accelerometer signals. During gimbaling, checks are made to assure proper clearance between the stage tail section and engines.

Instrumentation Compatibility — This test assures compatibility between the stage measuring instruments and the telemetry systems. Each telemeter channel is tested to assure that a linear output response of proper level can be obtained.

Simulated Flight — All vehicle systems are evaluated as a unit. The stage is programmed through a complete preflight and flight sequence during which time the umbilicals are retracted, the hydraulic system is activated, engine gimbaling tests are performed, and inboard and outboard engine cutoff sequences are verified.

S-IB STAGE SYSTEMS DESCRIPTIONS

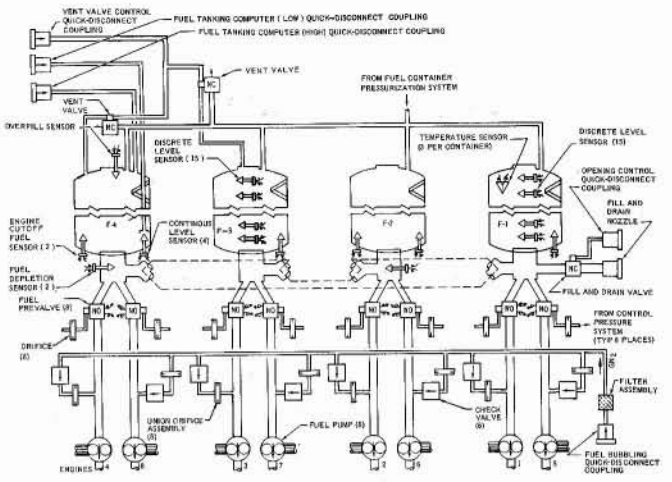
Fuel System

The stage fuel system receives RP-1 fuel from a ground source, stores the fuel, and then supplies it to the eight H-1 engines. The system consists of four fuel containers, pressurization components, distribution manifolds, control valves, switches, sensors, piping, interconnect lines and the connecting hardware required to fill or drain the containers, bubble the fuel before flight, pressurize the containers, and supply the fuel to the engines.

Equal pressurization of the containers and uniform distribution of fuel to the engines is maintained through interconnect lines at the top and bottom of the containers. In the event of an engine failure, the fuel normally consumed by the inoperative engine is supplied to the operating engines.

Each container supplies fuel to one outboard and one inboard engine through suction lines connected to the container sump. Two engine cutoff fuel sensors generate a signal when the fuel is decreased to their level that will initiate inboard engine shutdown. Similar engine cutoff sensors in the LOX system will initiate inboard engine shutdown if LOX depletion occurs prior to fuel depletion.

Outboard engine shutdown occurs approximately 3 seconds after inboard engine shutdown. The outboard engines are normally shut down when engine thrust decay, resulting from LOX depletion, causes the outboard engine thrust OK pressure switches to deactuate. The outboard engines are shut down simultaneously since the thrust OK pressure switches on all outboard engines are



CC-86

Fuel System Schematic

interconnected at this time. However, if fuel depletion occurs prior to LOX depletion, the outboard engines will be shut down when fuel reaches the level of fuel depletion sensors located in the container sump.

Probes in two of the four fuel containers permit telemetric monitoring of the fuel level during flight.

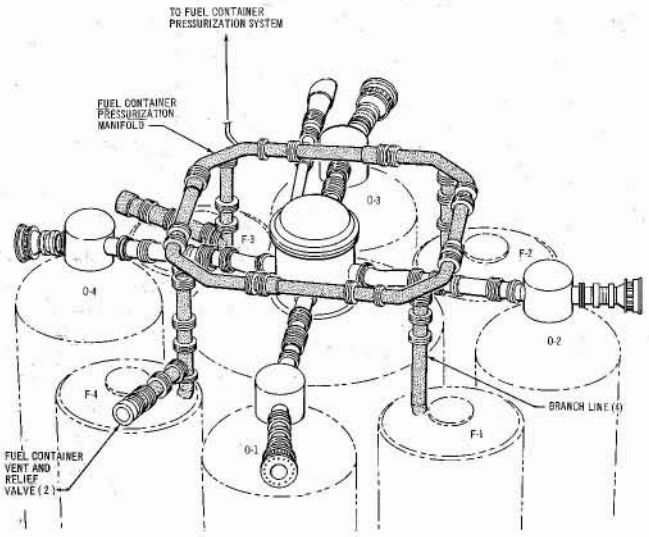
FUEL FILL

Prior to filling the containers with fuel, normally closed vent valves and a fuel fill and drain valve are opened. The vent valves are actuated by control pressure from a ground source supplied through the vent valve control quick-disconnect coupling. The fill and drain valve is actuated by control pressure from the ground source supplied through the opening control quick-disconnect coupling. Fuel is then pumped through the fill and drain nozzle into the sump of container F-1. The sumps of the containers are interconnected to ensure equal distribution of fuel. The fuel containers are initially filled to a predetermined level based upon a nominal density.

The rate of flow to the containers is controlled by a fuel tanking computer in the ground control station. The computer shuts off the supply when the containers are filled to the predetermined level. If the computer should malfunction and not shut off the fuel supply, an overfill sensor will initiate a signal to stop the fill sequence.

FUEL LEVELING

Fuel in the containers is maintained at the required level by adding or draining. The level is adjusted according to calculations by the fuel tanking computer based on pressure differentials in container F-4 and on fuel density calculations based on temperature measurements from each container. Temperature sensors monitor the temperature of the



Fuel Container Pressurization Lines

fuel in each container for the density calculations and electrically transmit the results to the computer.

FUEL DRAIN

In the event that fuel must be drained from the system, the containers are pressurized from a ground source. The vent valves are closed, the fuel fill and drain valve is opened, and fuel under pressure flows from the containers through the open valve and the fill and drain nozzle to the ground storage tank. Container pressure for the draining operation is maintained between 29.6 psia and 32.4 psia.

SUCTION LINES

Eight suction lines supply fuel from the container sumps to the engine turbopumps. Normally open fuel prevalves, located near the top of each suction line, may be closed by pneumatic pressure supplied by the stage control pressure system or by pneumatic pressure supplied from a ground source and through an orifice.

The fuel containers are interconnected at the top through the fuel container pressurization manifold. The fuel vent valves are connected into this manifold. The fuel containers are also interconnected at the sumps through interconnect lines.

An antivortex device and screen is located at the aft bulkhead of each container.

FUEL BUBBLING

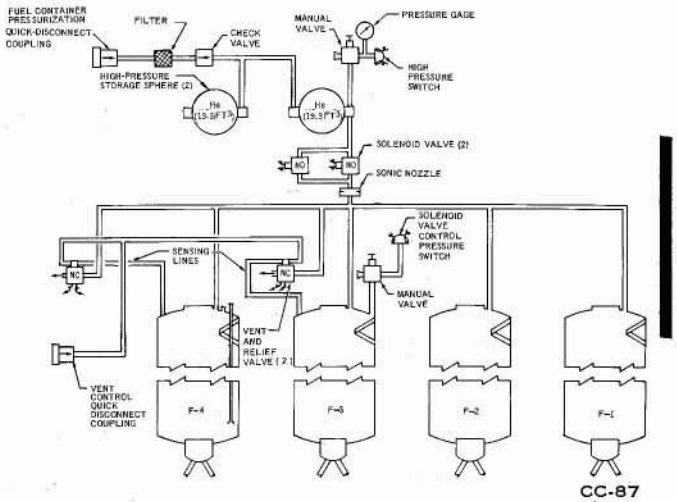
Pressurized GN_2 from a ground source is bubbled through each fuel suction line to agitate the fuel and thereby maintain a uniform fuel temperature throughout each container. Fuel bubbling begins just before LOX fill and continues until the start of fuel container pressurization.

FUEL CONTAINER PRESSURIZATION

The fuel containers are pressurized with helium starting from approximately two minutes and thirty-three seconds prior to launch and continuing until the S-IB flight is completed. The container pressure maintains a pressure head for the engine fuel pumps and provides structural integrity by preventing the formation of a vacuum in the containers as fuel is depleted during flight. Major components of the pressurizing system are high-pressure helium storage spheres, solenoid valves, pressure switches, a sonic nozzle, and distribution lines.

Prior to launch, two 19.3-cubic-foot, high-pressure storage spheres are pressurized to 3,000 psig. Helium is supplied from the launch facility through the fuel container pressurization quick-disconnect coupling connected to one of the launch facility umbilicals. The helium passes through a filter and a check valve before it enters the storage spheres. From the storage spheres, helium flows through two normally open solenoid valves, a sonic nozzle, the distribution line, and into the fuel containers. The sonic nozzle restricts the flow of helium. Prior to flight, container pressure is maintained at between 29.6 and 32.4 psia by a control pressure switch located at the top of container F-3. The switch controls the operation of the normally open solenoid valves and shuts off the helium supplied to the container when the pressure reaches 32.4 psia. If container pressure should exceed 35.7 psia, the sensing line will monitor the overpressurization and cause the vent valve to open. The containers are vented until normal pressure is maintained again.

A high pressure switch located on the storage spheres monitors sphere pressure and is a part of



Fuel Container Pressurization Schematic

the engine ignition interlock. The switch must indicate a minimum pressure of 2,800 ±50 psig or the countdown is stopped.

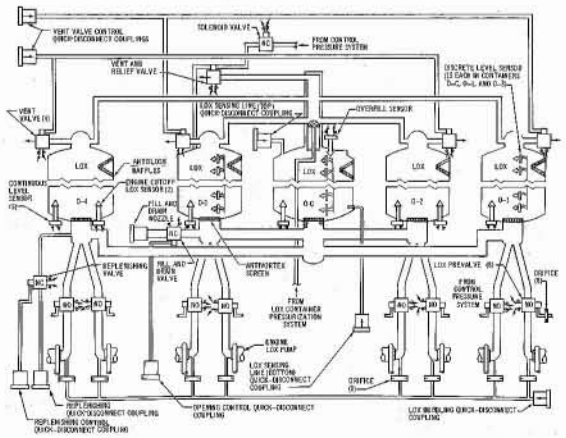
At liftoff, the electrical circuit to the solenoid valves and switches is disconnected and uninterrupted pressure from the spheres flows through the open solenoid valves and sonic nozzle into the containers. Engine fuel consumption is such that the system cannot be overpressurized and the pressure will fluctuate between a 17 psig upper limit and an 11 psig lower limit until stage burnout.

LOX System

The LOX system receives LOX from a ground source, stores the LOX in containers, and then supplies LOX to the eight H-1 engines. Major components of the system are the four outer LOX containers 0-1, 0-2, 0-3, and 0-4, the center LOX container 0-C, control valves, interconnect lines, suction lines, piping, switches, manifolds, sensors, vent valves, vent and relief valve, and the GOX flow regulator. These components provide the means of filling or draining the containers, replenishing LOX, bubbling LOX, pressurizing the containers, and supplying LOX to the H-1 engines.

Equal pressurization of the containers and uniform distribution of LOX to the engines is maintained through interconnect lines at the top and bottom of the containers. In the event of an engine failure, the LOX normally consumed by the inoperative engine is supplied to the operating engines.

The center container sump is connected to the sumps of the outer containers. Each outer container supplies LOX to one outboard and one inboard engine through suction lines connected to the container sump. When LOX falls to the level of the engine cutoff LOX sensors located in the bottom of containers 0-2 and 0-4, a signal is generated that initiates the inboard engine shutdown sequence, provided that similar sensors in the fuel system have not already initiated the sequence. Outboard engine shutdown occurs approximately three seconds after inboard engine shutdown. The outboard engines are normally shut down when engine thrust decay, resulting from LOX depletion, causes the outboard engine thrust OK pressure switches to deactuate.



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LOX System Schematic

The outboard engines are shut down simultaneously since the "thrust O.K." pressure switches on all outboard engines are electrically interconnected at this time. However, if fuel depletion occurs prior to LOX depletion, the outboard engines will be shut down when fuel reaches the level of either of two depletion sensors located in the sumps of fuel containers F-2 and F-4. Sensors in three LOX containers permit telemetric monitoring of the LOX level.

LOX FILL

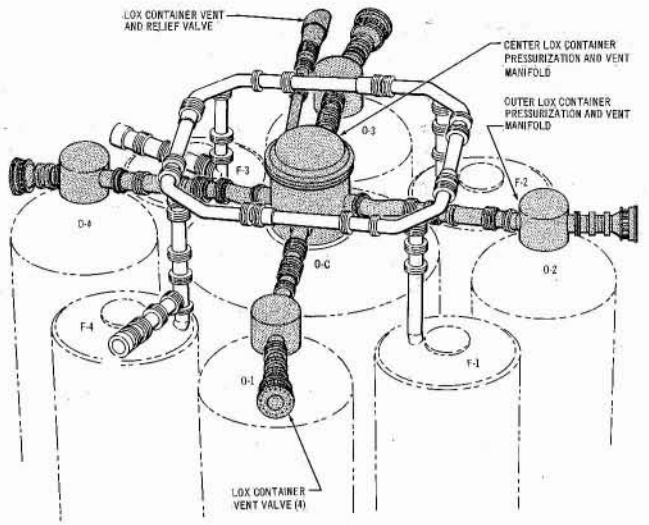
Before filling the containers with LOX, the four normally closed vent valves and the vent and relief valve are opened. GN_2 control pressure from the launch facility opens the vent valves. The vent and relief valve is opened by GN_2 pressure from the onboard control pressure system through the solenoid valve.

After the vent valves are opened, the normally closed fill and drain valve is opened by control pressure supplied by the ground control station. LOX is then pumped through the fill and drain nozzle and the open fill and drain valve into the sump of container 0-3. The sumps of the containers are interconnected to ensure equal distribution of LOX. The LOX containers are initially filled to a level based on nominal fuel density.

The flow into the containers is controlled by a ground station LOX tanking computer. When the containers are filled to the predetermined level, the computer shuts off the LOX supply. If the computer should malfunction and not shut off the LOX supply, an overfill sensor will generate a signal to stop the fill sequence.

LOX REPLENISHING

LOX in the containers must be continuously replenished until containers are pressurized to compensate for boil-off losses and changes in fuel density. Corrections are applied to the LOX tanking computer and the containers are replenished from the ground LOX storage tank. A normally closed replenishing valve is opened by control pressure applied from the launch facility. The LOX flows into the sump of container 0-4. Differential pressure is sensed by the top and bottom sensing lines in container 0-C and routed to the tanking computer through the LOX sensing line quick-disconnect coupling.



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LOX Container Pressurization Lines

LOX DRAIN

In the event that LOX must be drained from the system, the vent valves and vent and relief valves are closed, the containers are pressurized, and the fill and drain valve is opened. LOX under pressure then flows from the containers through the open valve and fill and drain nozzle to the ground storage tank.

SUCTION LINES

Eight suction lines conduct LOX from the outer container sumps to the engine turbopumps. Normally open LOX prevalves are located near the top of each suction line and may be closed by pneumatic pressure from the stage control pressure system or the launch facility should the need arise prior to launch. The prevalves also provide a backup to the main LOX valve in the H-1 engines for LOX shutoff.

The LOX containers are interconnected at the top by lines between the LOX pressurization and vent manifold and each outer container. Interconnect lines connect the sump of the center container to the sumps of each outer container.

LOX BUBBLING

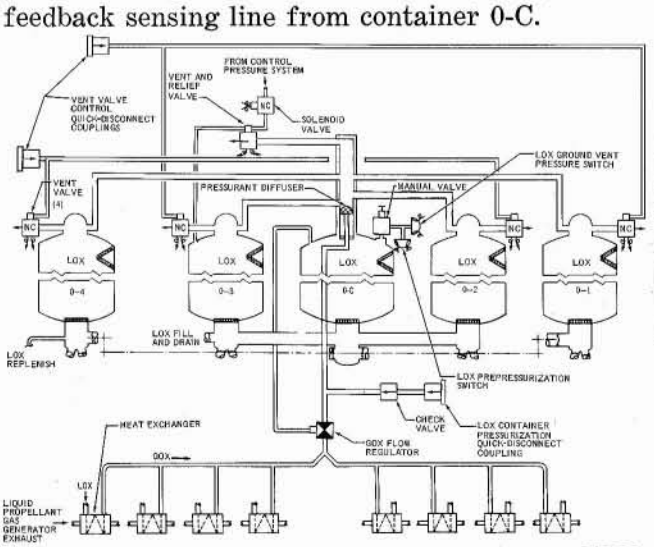
Pressurized helium is applied at the inlet of each engine LOX pump and bubbles up through the LOX suction lines into the container. The helium rising through the LOX stabilizes the temperature at the pump inlet to prevent pump cavitation. LOX bubbling is initiated just after the initiation of fuel container pressurization and continues until LOX container pressurization occurs.

LOX CONTAINER PRESSURIZATION

The LOX containers are pressurized with helium during preflight operations to provide a pressure head at the inlet of the engine LOX pumps. During flight, container pressure also provides structural integrity to the containers. GOX converted from LOX by the heat exchanger on the H-1 engines is used as the pressurant for inflight operation.

To pressurize the LOX containers before engine ignition, the vent and relief valve and the vent valves are closed. Helium flows from the launch facility through the LOX container pressurization quick-disconnect coupling, a check valve, and a pressurant diffuser into container 0-C. The helium then flows from container 0-C through the interconnecting lines to the outer containers, thus equally pressurizing all containers. The LOX prepressurization switch monitors container pressure and shuts off the helium supply when the pressure reaches 57.7 psia. In the event the switch fails, the vent and relief valve will open when pressure in the sensing line between the valve pilot and container 0-3 increases to between 60 and 63 psia. However, if the vent and relief valve should fail to open by this method, the LOX ground vent pressure switch will actuate at 67.5 psia to energize the solenoid valve and open the vent and relief valve. The manual valve is used to calibrate the pressure switches.

At liftoff, the LOX ground vent pressure switch is disabled and overpressurization protection is provided by the vent and relief valve. Helium from the launch facility is disconnected and GOX from the H-1 engine heat exchanger assemblies is used as the pressurant. The GOX output from all eight heat exchanger assemblies flows into a manifold, through the GOX flow regulator, the pressurant diffuser in container 0-C, and through the interconnecting lines to the outer containers. The GOX flow regulator maintains LOX container pressure at approximately 50 psia. The regulator is controlled by a pressure feedback sensing line from container 0-C.



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LOX Container Pressurization Schematic

The LOX prepressurization switch, which is also disabled at liftoff, is enabled again at liftoff +30 seconds. During flight, the switch continues to monitor LOX container pressure and transmits a signal to the solenoid valve to open the vent and relief valve should container pressure increase to 58.5 psia.

Control Pressure System

Major components of the system are a storage sphere, a pressure regulator, a manifold, a relief valve, switches, solenoid-operated valves, filters, and monitoring devices.

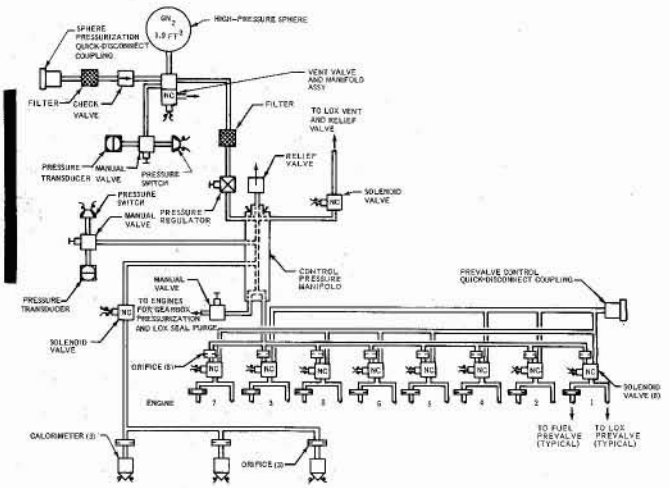
The control pressure system receives pressurized GN_2 from the launch facility, stores it, and then supplies it at a reduced and regulated pressure to operate pneumatic valves in the fuel and LOX systems, open LOX vent valves, purge calorimeters, pressurize the engine gearboxes, and purge the LOX seal area of the engine turbopumps.

The high-pressure storage sphere is pressurized from the launch facility through the sphere pressurization quick-disconnect coupling, a filter, a check valve, and the vent valve and manifold assembly. The 1.0-cubic foot storage sphere is charged to 3,000 psig; a pressure switch and pressure transducer monitor sphere pressure. When necessary, the system is vented through the vent valve and manifold assembly.

High-pressure GN_2 flows from the storage sphere through the vent valve and manifold assembly and the filter to a pressure regulator. The pressure regulator reduces and regulates the pressure to 750 psig and supplies the gas to a control pressure manifold. Pressure in the manifold is monitored by a pressure switch and pressure transducer. The manifold is protected from overpressurization by a relief valve.

Control pressure GN_2 is routed from the manifold to a solenoid valve for the operation of the LOX vent and relief valve. Control pressure GN_2 is also routed to eight solenoid valves to close the LOX and fuel prevalves at the termination of static test and during the inflight engine shutdown sequence at stage burnout. A launch facility connection through a quick-disconnect coupling allows ground control of the prevalves prior to launch.

Control pressure GN_2 is also used to pressurize the engine gearboxes and purge the LOX seals on the turbopumps; a manual valve provides shutoff control. GN_2 supplied through a solenoid valve is used to purge the windows of three calorimeters so that combustion products do not accumulate on the windows and cause the calorimeters to give erroneous indications.



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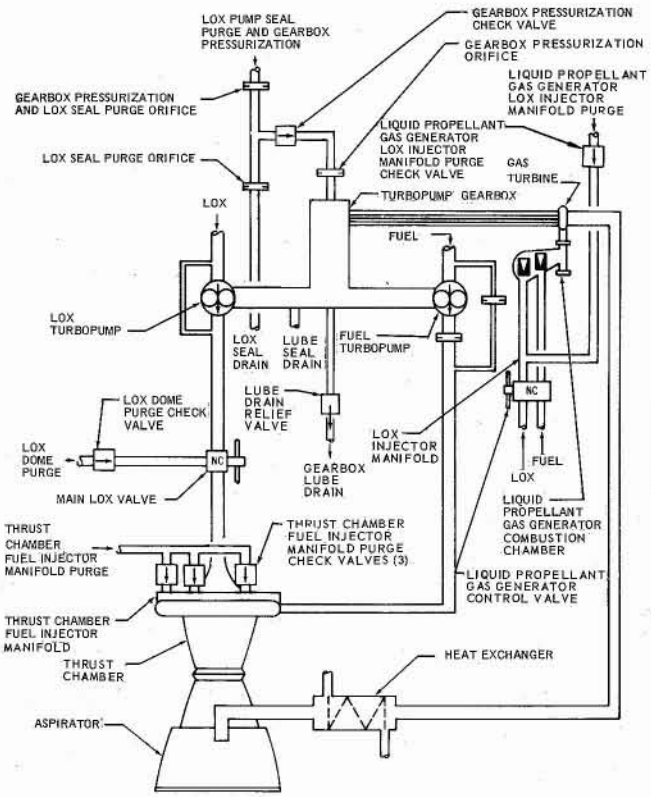
Control Pressure System Schematic

Engine Purge and Gearbox Pressurization System

Several GN_2 purges are started during launch preparations to prevent contamination of the engine and reduce the possibility of an accidental fire before engine ignition.

LOX PUMP SEAL PURGE AND GEARBOX PRESSURIZATION

The LOX pump seal purge and gearbox pressuri-



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Engine Purge and Gearbox Pressurization System

zation operation are described together since both are supplied GN_2 from a common source.

The gas is supplied by the vehicle control pressure system; the supply is routed through a ring-line manifold with branch lines to each engine gearbox. The LOX pump seal purge and gearbox pressurization starts when the S-IB stage control pressure system is pressurized and continues throughout launch preparations, engine starting, and S-IB flight. If the launch is cancelled, purging continues until all LOX within the engine LOX pump has boiled off.

Low-pressure GN_2 purge is applied to the space between the LOX and lube seals in the turbopump to isolate LOX leakage from the lubricant leakage. LOX and lubricant leakage into the area between the seals is drained overboard through separate drains.

Gearbox pressurization is accomplished with GN_2 also reduced in pressure. The check valve and the relief valve operate together to maintain a pressure within the gearbox to prevent the lubricant from foaming at high altitudes. The relief valve also functions to bleed spent lubricant (RP-1 fuel and Oronite 262 additive) to the atmosphere.

The pressure within the gearbox also forces any fuel that leaks past the fuel seals and into the gearbox to drain overboard through the gearbox lube drain. The leakage can be visually monitored prior to engine ignition.

LOX DOME PURGE

The LOX dome of each H-1 engine is purged by both a low-level purge and a full-flow purge. The low-level purge maintains a slight positive GN_2 pressure in the LOX dome to prevent contaminants from entering the thrust chamber nozzle and flowing to the injector plate and the LOX dome. It also prevents moisture from condensing in the area. The low-level purge is started prior to propellant loading and continues until shortly before engine ignition; GN_2 pressure and flow rate are then increased to the full-flow level. The full-flow purge continues until LOX pressure in the LOX dome is greater than purge pressure. If a launch is cancelled, the full-flow purge resumes as LOX pressure decays below purge pressure. The GN_2 purge then expels LOX from the LOX dome and the LOX bootstrap line. After a short interval, the full-flow purge rate is reduced to the low-level rate.

Ground source GN_2 for both purge levels flows through the LOX dome purge check valve, through a manifold on the main LOX valve, and then into the LOX dome. The GN_2 is vented through the thrust chamber nozzle.

THRUST CHAMBER FUEL INJECTOR MANIFOLD PURGE

The thrust chamber fuel injector manifold purge prevents LOX from entering the fuel injector manifold during engine ignition. The purge is started approximately 30 seconds before engine ignition. Ground source GN_2 flows through a ring-line manifold around the engine compartment and then into a purge manifold of each of the engines. Each purge manifold distributes the GN_2 into three thrust chamber fuel injector manifold purge check valves. The GN_2 flows through the thrust chamber fuel injector manifold, through the injector plate, and out the thrust chamber nozzle. The purge is stopped when fuel pressure builds up in the fuel injector manifold, as a result of engine starting, and closes the three check valves.

LIQUID PROPELLANT GAS GENERATOR LOX INJECTOR MANIFOLD PURGE

The liquid propellant gas generator LOX injector manifold purge prevents the combustion products from the solid propellant gas generator from contaminating the liquid propellant gas generator LOX injector manifold. The purge is started just before engine ignition and is stopped by LOX pressure buildup in the manifold. If a launch is cancelled, the purge commences immediately following engine shutdown and continues until the solid propellant gas generator has been removed.

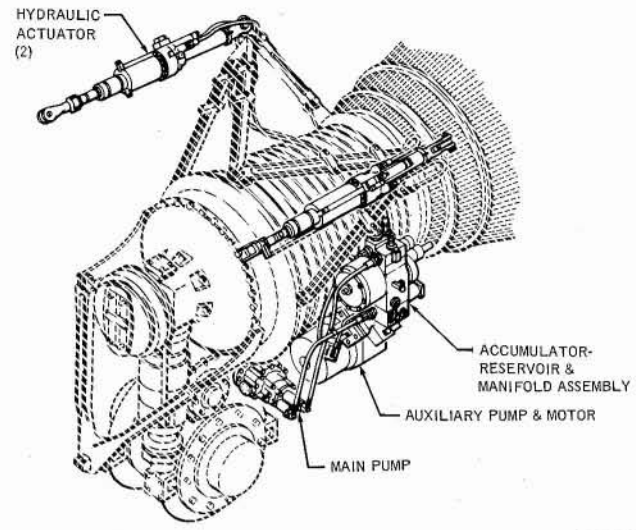
Ground source GN_2 flows through branch lines leading to each engine, and through a check valve into the LOX injector manifold. The GN_2 purge then flows through the liquid propellant gas generator, the gas turbine, and the heat exchanger. The GN_2 purge is exhausted through the engine aspirator.

Hydraulic System

Each outboard engine has an independent, closed-loop hydraulic system that gimbals the engine for vehicle flight control, moving the engine in proportion to the magnitude of an electrical input signal. Movement is provided by two hydraulic actuators that may be extended or retracted independently or simultaneously.

Major components of each system are a main hydraulic pump, an auxiliary hydraulic pump and motor, an accumulator-reservoir and manifold assembly, and the hydraulic actuators.

The accumulator is filled with GN_2 from a ground source through the GN_2 charging valve. Hydraulic fluid pumped from the ground source flows through the high-pressure quick-disconnect coupling and the main filter into the reservoir and the



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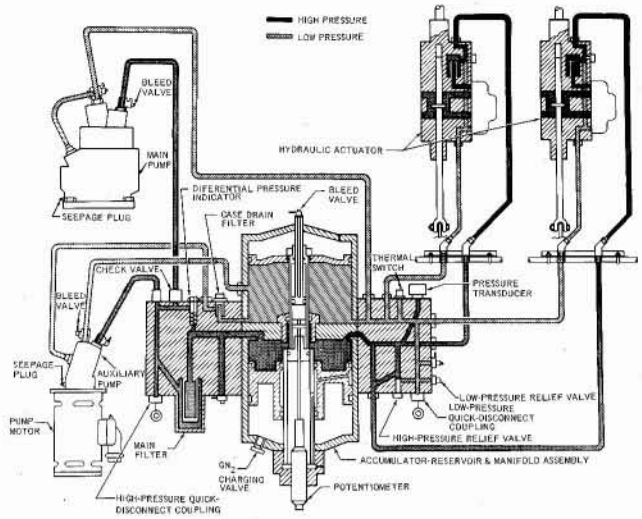
Hydraulic System Components



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Hydraulic System Schematic

system. Fluid level in the reservoir is indicated by a potentiometer. Any excess hydraulic fluid returns to the ground source through the low-pressure quick-disconnect coupling during servicing.

An electric auxiliary pump is used for ground checkout of the hydraulic system operation. A check valve protects the main pump from high-pressure fluid during auxiliary pump operation. After engine ignition, a check valve protects the auxiliary pump from high pressure during main pump operation.

The main pump is driven by the turbopump after engine ignition. Hydraulic fluid is drawn from the reservoir portion of the accumulator-reservoir and manifold assembly, and pumped through a check valve and the main filter into the accumulator portion of the accumulator-reservoir and manifold assembly. Pressure is increased to approximately 3,200 psig on the high pressure side of the main pump. Fluid flows from the accumulator-reservoir and manifold assembly to the hydraulic actuators.

An electro-hydraulic servovalve in each actuator directs high-pressure fluid against the required side of the actuator piston, when activated by an electrical signal. The piston is connected to the actuator arms, which extend or retract, and provide gimbaling action to the engine. Fluid flows from the hydraulic actuators back to the reservoir.

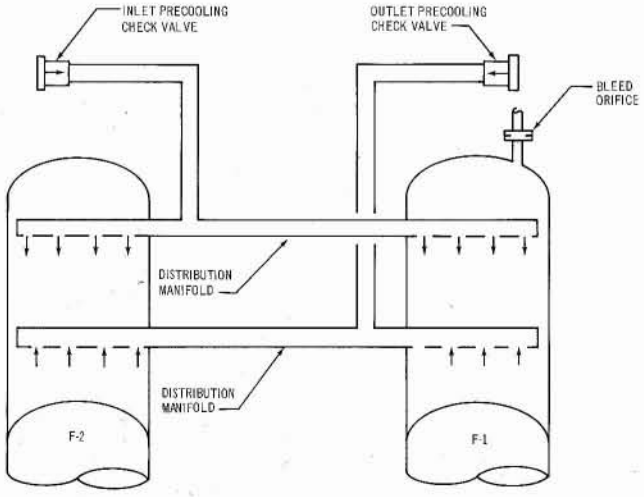
Instrument Compartment Environmental Conditioning System

The instrument compartments located in the top skirt of fuel containers F-1 and F-2 are environmentally conditioned prior to flight. Conditioned air or GN_2 supplied from the launch facility environmental conditioning system is ducted into the instrument compartments to dissipate heat generated by the electronic instruments, and reduce the possibility of fire during launch.

Compartment conditioning is initiated when the electronic equipment is turned on at approximately T minus 19.5 hours. Air is used until LH_2 tanking is initiated in the S-IVB stage; GN_2 is then used for the remainder of the conditioning period until liftoff. At liftoff, the precooling check valves close and a bleed orifice prevents an increase in compartment pressure during flight.

Tail Unit Conditioning and Water Quench System

The tail unit conditioning and water quench system is provided to direct conditioned air, conditioned GN_2, or water into the engine compartment. The conditioned air or GN_2 is supplied by the launch complex to provide a temperature controlled atmosphere in the tail unit area. In the



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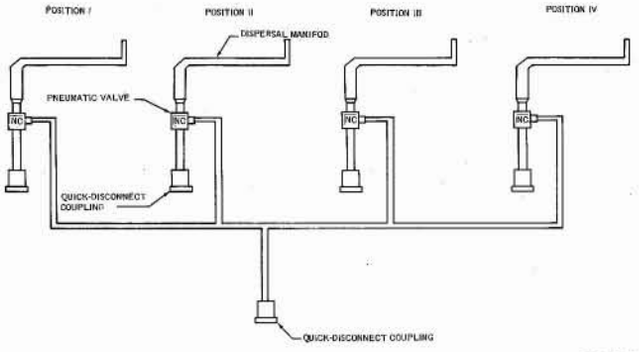
Instrument Compartment Environmental Conditioning System



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Boattail Conditioning and Water Quench System Schematic

event of fire in the engine compartment, or any malfunction which causes engine shutdown, a cold GN_2 deluge and, if necessary, a water quench is supplied to the tail unit area. All mediums flowing through the system use the same vehicle network of valves, pipes, and dispersal manifolds.

At liftoff, the system is separated from the ground source; therefore, tail unit conditioning and fire protection are not provided during flight.

TAIL UNIT FIRE DETECTION SYSTEM

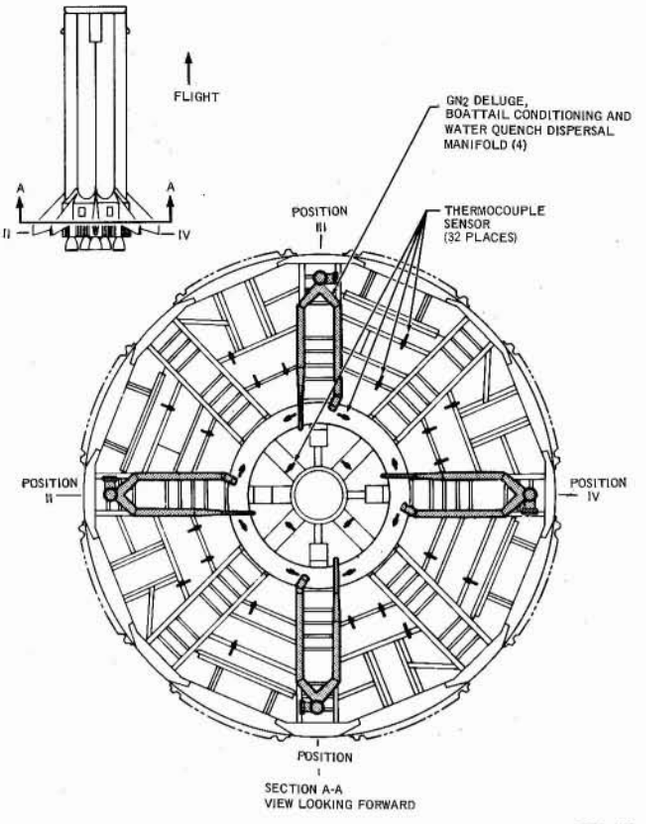
A fire detection system is provided to detect any fire that may occur in the engine compartment prior to liftoff. The fire detection system consists of 32 thermocouple sensors located on the aft thrust structure and the firewall substructure. The sensors are connected in four loops of eight sensors. The eight sensors in each loop are positioned at a similar location near each engine; thus, the temperature rise-rate in four critical areas of each engine is monitored until liftoff.

If a sensor detects a fire prior to engine ignition, a signal is transmitted to the launch control center to initiate a cold GN_2 deluge of the engine compartment. If a fire is detected during the time between engine ignition and launch commit, the signal initiates the engine cutoff sequence and the cold GN_2 deluge.

If the cold GN_2 deluge does not extinguish the fire, a water quench operation is manually initiated from the launch control center. The GN_2 deluge uses the same dispersal manifold network as the tail unit conditioning and water quench system.

Range Safety System

Each vehicle launched from the ETR must have two separate and independent methods of emergency flight termination for use if the vehicle should become a safety hazard during powered flight. Air Force Missile Test Center safety regulations require missiles using liquid propellants to have systems that accomplish zero thrust and propellant dispersion.



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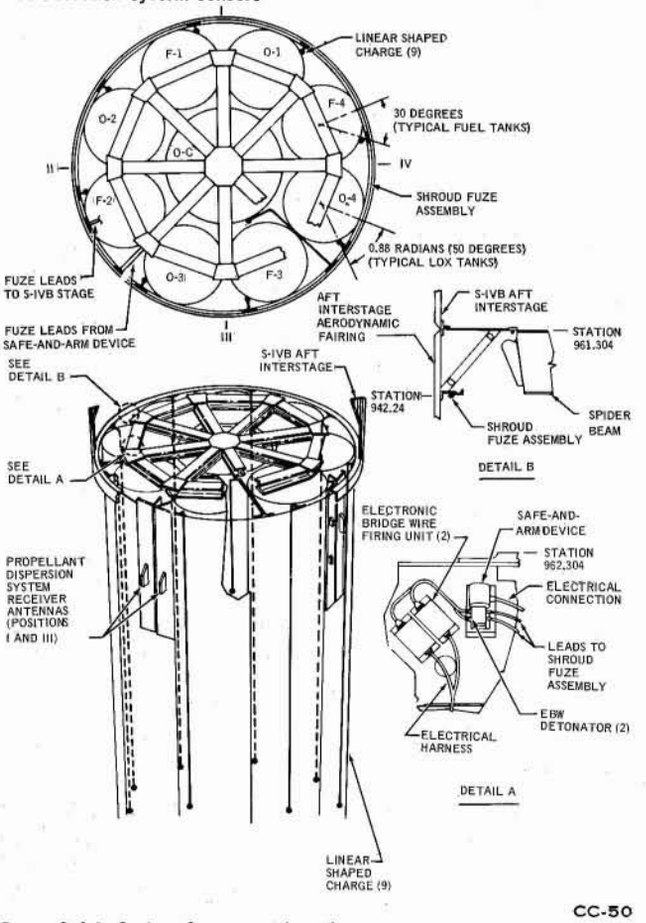
Fire Detection System Sensors



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Range Safety System Component Locations

The ETR Safety Officer decides whether or not a safety hazard exists and bases his decision on (1) the imminence of an explosion, and (2) deviation of the vehicle from the programmed flight path.

The S-IB stage range safety system consists of four receiver antennas, two command receivers, two command controllers, two electronic bridgewire firing units, two EBW detonators, and a safe-and-arm device. The safe-and-arm device connects the system to the shroud fuze assembly and nine linear-shaped charges. Reliability is further increased by supplying power to each of the dual sets from separate batteries.

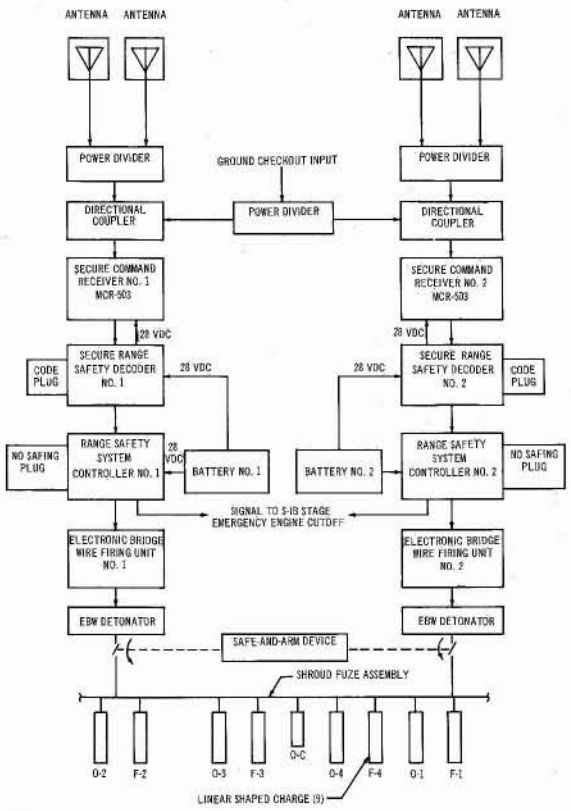
A ground-based command transmitter sends the frequency-modulated audio-tone coded signals that initiate engine cutoff, charge the capacitor in the electronic bridgewire firing unit, and trigger the propellant dispersion system. Four receiver antennas mounted in sets of two on opposite sides of the stage, receive the signals. The signals are demodulated within the command receivers. The resulting audio tones are applied to the decoder, which separates the tones according to frequency, and energizes relays that generate the engine cutoff signal, the arming command, and the destruct command.

The command controller directs signals from the command receiver to the proper units and relays supervisory signals to the ground station. Electrical connectors are provided for attaching a no-destruct-delay plug or a delay timer to the controller. The no-destruct-delay plug completes the electrical circuit immediately when no time delay is desired. A delay timer will delay the destruct command to allow a payload to be jettisoned before the vehicle is destroyed. The destruct command, transmitted approximately 3 seconds after the arming command, is coupled to the electronic bridgewire firing unit to fire the EBW detonator.

The EBW detonator is coupled to the shroud fuze assembly through the safe-and-arm device. The shroud fuze assembly interconnects the nine shaped charges on the nine propellant tanks.

When the system is actuated, the linear-shaped charges fire and the propellant tanks are split open, resulting in complete propellant dispersion and destruction of the vehicle.

The safe-and-arm device provides safety to personnel during the installation of the EBW detonators and the fuze assembly. The device is a rotary mechanical block that, when in the safe position, blocks the flame path between the EBW detonators and the shroud fuze assembly. Prior to liftoff, the safe-and-arm device can be armed or disarmed



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Range Safety System Block Diagram

electrically or manually. Prior to liftoff, the device is actuated to the armed position.

Electrical System

Two independent 28-volt bus systems distribute the power throughout the stage. Generally, constant loads are supplied from one bus and variable loads are supplied from the other bus. However, exceptions to this general rule are frequently made to form redundant circuits in the interest of reliability.

Prior to launch, primary power is supplied by the launch complex through umbilical connections. After launch, two wet-cell storage batteries located in instrument compartment No. 2 furnish 28 vdc. Actual power transfer occurs approximately 25 seconds before engine ignition. Two squib-actuated switch assemblies are actuated at liftoff to complete power transfer circuits which parallel the normal transfer circuits. This ensures that the stage is switched from ground power to battery power, compensates for possible contactor bounce, and makes the transfer non-reversible and virtually failure proof.

The 28 vdc power is switched and distributed to the inflight subsystems through the power distributor, also located in instrument compartment No. 2.

Many of the transducers in the measurement system require 5 vdc for excitation. Three master measuring voltage supplies are installed on the stage to convert 28 vdc to precisely regulated 5 vdc.

CONTROL COMPONENTS

A switch selector is installed on the S-IB stage to connect and disconnect the various flight sequence commands during S-IB stage powered flight.

The switch selector installed in instrument compartment No. 2 controls time-referenced events in the S-IB stage. It receives digital-coded commands from the data adapter in the Instrument Unit, decodes the commands, and transmits them to the main distributor.

Flight Measurement Program

The requirements for preflight and inflight performance measurements of the Saturn vehicles differ substantially from those of conventionally guided missile systems. A large number of measurements must be obtained to meet the stringent demands of the Saturn research and development program. The various physical events and environmental conditions which prevail throughout the vehicle before and during flight must be made available to ground stations in a precise, real-time format.

About 440 measurements are made and telemetered during the S-IB stage flight. Before launch, approximately 110 measurements will be transmitted to the blockhouse by hardwire connections.

Many of the source signals are not suitable for direct transmission by the telemetry system; therefore, signal conditioning devices are required to modify the signals. The conditioning devices are replaceable modules installed in 8 measuring racks in the tail unit area and in 3 measuring racks in instrument compartment No. 2.

Measuring distributors are junction boxes which connect the measurement signals to the telemetry systems and provide points for checkout, maintenance, and modification of the systems. Three distributors are located in the tail unit area and one is located in instrument compartment No. 2.

TELEMETRY SYSTEMS

Stage performance measuring signals, when grouped according to frequency and accuracy requirements, can be most effectively transmitted by using several types of telemeters. Four telemeter systems are required to transmit the S-IB stage measuring signals. Most of the components of the telemetry systems are located in instrument compartment No. 1, however, a telemetry system multiplexer is installed in the aft skirt of fuel containers F-1 and F-2. The telemeters transmit data through a common antenna system.

Telemeters F1 and F2

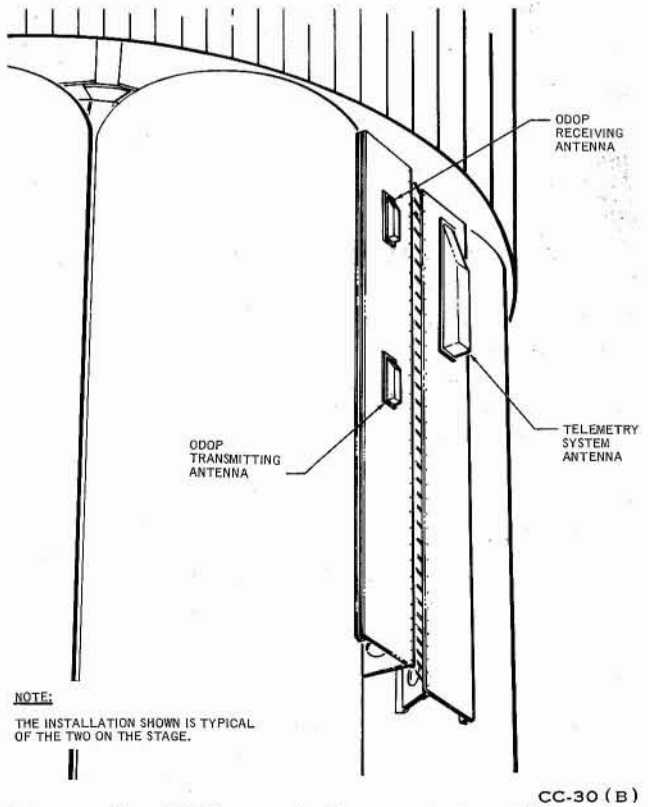
Telemeters F1 and F2 are identical systems which transmit narrow band, frequency-type data such as that generated by strain gages, temperature gages, and pressure gages. The system can handle 234 measurements on a time-sharing basis and 14 measurements transmitted continuously. Data may be sampled 120 times per second, or 12 times per second.

Telemeter S1

Telemeter S1 transmits wide band frequency-type data generated by vibration sensors. The S-IB stage measuring program requires 39 data sources transmitted on a 25 per cent duty cycle, 8 data sources transmitted on a 50 per cent duty cycle, and 4 data sources transmitted continuously.

Telemeter P1

Telemeter P1 transmits pulse code modulated, or "bang-bang" type data. This type of data is generated by limit switches, pressure-actuated switches, valves, and relays. Five multiplexers supply data to the telemeter; three handle 234


ODOP RECEIVING ANTENNA
ODOP TRANSMITTING ANTENNA
TELEMETRY SYSTEM ANTENNA
NOTE:
THE INSTALLATION SHOWN IS TYPICAL OF THE TWO ON THE STAGE.

CC-30 (B)

Antennas — The ODOP transponder has separate transmitting and receiving antennas. The two antennas are located on the antenna panel at stage position IV.

bits of digital data each while two multiplexers have a capacity of 100 data inputs.

Telemetering Calibrator

A telemetering calibrator installed in instrument compartment No. 1 improves the accuracy of the telemetry systems. The calibrator supplies known voltages to the telemeters periodically during the S-IB stage operation. Their reception at tracking stations provides a valid reference for data reduction.

Tape Recorder

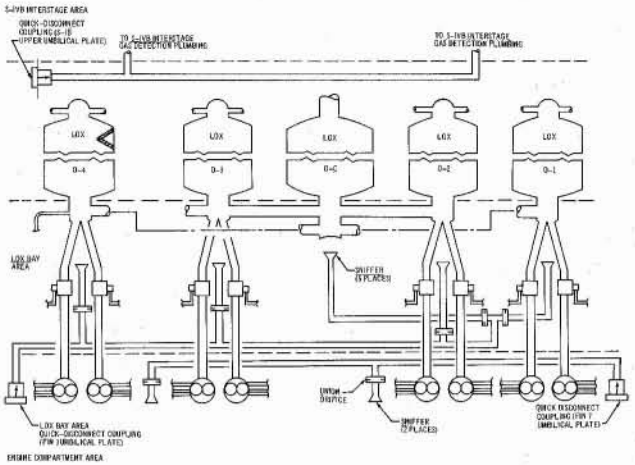
The effects of retrorocket firing attenuation can seriously degrade the telemetry transmission during stage separation; therefore, a tape recorder installed in an instrument compartment records data for delayed transmission. The commands for tape recorder operation originate in the Instrument Unit.

Tracking System

The S-IB stage carries a transponder to facilitate ground tracking. The transponder, installed in instrument compartment No. 1, is part of the ODOP tracking system. The ODOP is an elliptical tracking system that measures the sum of the ranges between the stage and three ground stations. The range sum is determined by measuring the total Doppler shift in frequency of a continuous wave radio frequency. Since the transponder is phase-coherent, the Doppler shift is determined primarily by the range and velocity of the stage.

Hazardous Gas Detection System

Certain components in the S-IB stage and S-IVB stage propulsion systems of the Saturn IB vehicle are subject to some gas leakage. This leakage, although permissible to a degree, could create a flammable or explosive atmosphere in selected areas of the vehicle prior to liftoff.



CC-100

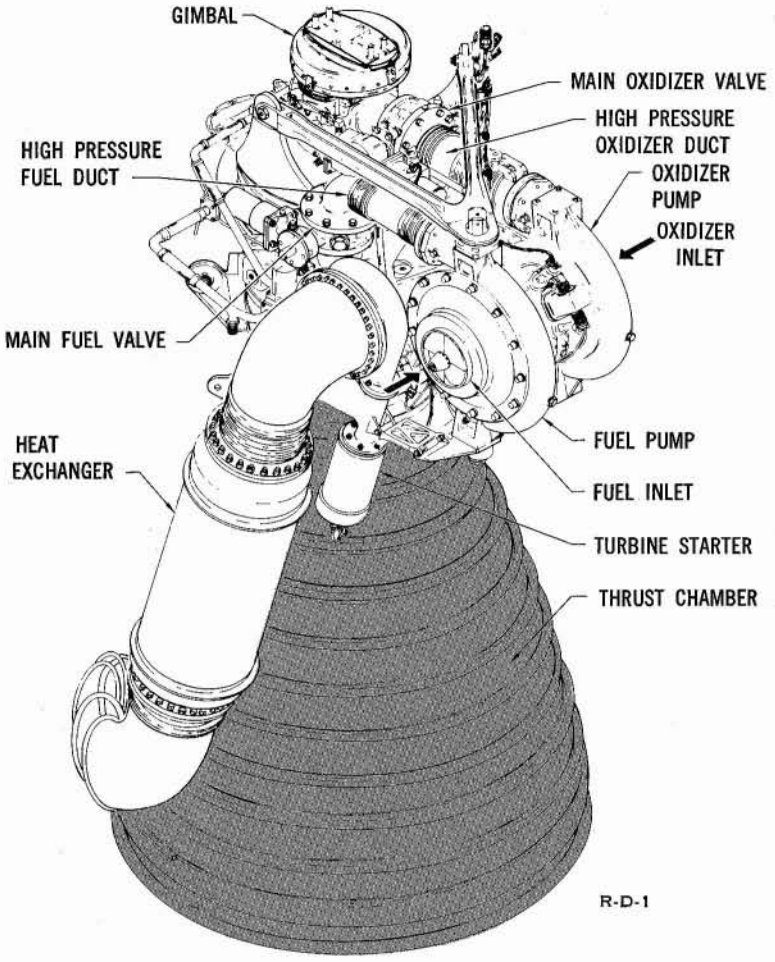
Hazardous Gas Detection System Schematic

The hazardous gas detection system monitors the leakage of hazardous gases in the fire and explosion sensitive areas of the vehicle by continuously sampling the atmosphere in these areas.

The system consists of plumbing lines located in the S-IVB interstage, the LOX bay area of the tail unit and the engine compartment. The lines contain orifices and open, flared-end "sniffer" tubes. The sniffers are positioned to sample the atmosphere of the explosion sensitive areas.

A vacuum pump in the ground complex draws air from the three sensitive areas through quick-disconnect couplings on swing arm No. 1 and two short cable masts. The sample of air thus obtained is drawn into a go/no go type of chemical sensitivity meter. The meter registers only when the volume of hazardous gas per cubic foot of air exceeds the threshold of fire or explosion capability.

H-1 ENGINE FACT SHEET



LENGTH	8 ft 6 in
WIDTH	5 ft 6 in
NOZZLE EXIT DIAMETER	3 ft 10 in ID
THRUST (at sea level)	200,000 lb
SPECIFIC IMPULSE	263 sec
RATED RUN DURATION	155 sec
FLOWRATE: Oxidizer	526 lb/sec (3330 gpm)
Fuel	236 lb/sec (2092 gpm)
MIXTURE RATIO	2.23:1 oxidizer to fuel
NOMINAL CHAMBER PRESSURE (P_c)	633 psia
WEIGHT, FLIGHT CONFIGURATION: Inboard	2,009 lb
Outboard	1,982 lb
EXPANSION AREA RATIO	8:1
COMBUSTION TEMPERATURE	6000°F

Note: H-1 engines will be uprated for vehicle SA-206 and all subsequent operational vehicles.

H-1 ENGINE

H-1 ENGINE DESCRIPTION

Each H-1 engine is rated at 200,000 pounds of thrust. The eight-engine cluster used on the S-IB stage produces a total of 1.6 million pounds of thrust. While these eight engines are basically the same design, there are important differences between the four inboard and the four outboard engines. Outboard engines are equipped with two component systems which are not employed on the inboard engines: an aspirator and gimbal outriggers. Components of the engine are mounted on a single, regeneratively-cooled thrust chamber of stainless steel, tubular wall construction.

H-1 ENGINE SYSTEMS DESCRIPTION

Major systems common to both inboard and outboard engines are: thrust chamber and gimbal assembly, exhaust system, gas generator and control system, propellant feed system, turbopump, fuel additive blender unit, and electrical system.

Thrust Chamber and Gimbal

The thrust chamber and gimbal assembly includes a gimbal, LOX dome, injector and hypergol container, and thrust chamber body. The thrust chamber receives the propellants under turbopump pressure, mixes and burns them, and imparts a high velocity to the expelled combustion gases to produce thrust for vehicle propulsion. The H-1 engine thrust chamber also serves as a mount or support for all engine and certain vehicle hardware.

GIMBAL

The gimbal secures the thrust chamber to the vehicle thrust frame and is mounted on the thrust chamber dome and elbow assembly. The gimbal is essentially a universal joint consisting of a cross-shaped unit incorporating bearing surfaces, upper and lower retainers, pillow blocks, and thrust vector aligning slides.

Purpose of the gimbal bearing assembly on the outboard engines is to permit a thrust chamber pivotal movement, which results in a maximum angle of 10 degrees from the normal (14 degrees when both actuators are extended) in the plane of the actuators as thrust vector control actuation movement is transmitted to the thrust chamber gimbal outriggers. The gimbal provides for positioning and thrust alignment capabilities.

OXIDIZER DOME

The oxidizer dome encloses the forward end of the thrust chamber combustion area, provides a mount for gimbal and thrust chamber assembly, and transmits engine thrust forces to the vehicle structure. Oxidizer from the turbopump enters the thrust chamber through the elbow of the dome assembly. The elbow is also the takeoff port of the gas generator oxidizer bootstrap line.

THRUST CHAMBER INJECTOR

The purpose of the injector is to meter the amount of propellants entering the thrust chamber combustion area and inject these propellants in a prescribed pattern, thereby ensuring satisfactory combustion. The injector incorporates 21 concentric circular passages (rings). Fuel and oxidizer are distributed through alternate rings; fuel flows through the outermost ring and in each alternate ring.

Propellants flow through angled orifices in the rings and into the thrust chamber causing the propellants to impinge in the combustion area in a like-on-like pattern (oxidizer-on-oxidizer and fuel-on-fuel).

Baffles, consisting of six copper fins and a copper hub, are mounted on the face of the injector. The hub is mounted over the fourth fuel ring. The six fins extend radially from the hub to the outer diameter of the injector face to divide the area equally. Holes drilled in the baffles mate with corresponding holes in the fuel and oxidizer rings allowing fuel and oxidizer to flow through the baffles into the combustion area. The baffled configuration results in increased combustion stability.

The welding of the hypergol container to the injector plate assembly has created the unitized injector and hypergol container. Pyrophoric fluid flows from the hypergol container through seven passages in the injector to the igniter fuel housings brazed into the face of the injector.

The injector is the primary thrust-experiencing component. Combustion zone pressure acts upon the injector face area, producing a thrust force which is transmitted from the injector to the oxidizer dome, to the gimbal, and subsequently, to the vehicle structure.

THRUST CHAMBER BODY

The thrust chamber body is a deLaval nozzle (100% bell) and consists of a combustion section, a converging throat section, and a diverging section through which combustion gases are expanded and accelerated.

The chamber body wall is constructed of longitudinal stainless steel tubes joined together by furnace brazing and retained by external stiffening

rings and tension bands. The tubes are of orthogonal cross section and are shaped to conform to thrust chamber contours. This type of construction permits thrust chamber regenerative cooling during engine operation by fuel flow through the tubes. A tapered fuel manifold gives equal flow to each down tube, providing a more even fuel flow through the thrust chamber cooling tubes. On inboard engines the thrust chamber body incorporates an improved design to accommodate the installation of the inboard turbine exhaust system.

Exhaust System

The engine exhaust system on the outboard engines consists of a turbine exhaust duct, heat exchanger, heat shield, and a heat exchanger LOX supply line. The thrust chambers on all outboard engines have an aspirator installed to distribute the exit flow of exhaust gases. Inboard engines have an engine-mounted turbine exhaust system consisting of a turbine exhaust hood, heat exchanger, turbine exhuast duct, heat shield, and heat exchanger LOX supply line.

TURBINE EXHAUST HOOD

The turbine exhaust hood is a stainless-steel, welded elbow assembly incorporating two mating flanges, two doubler rings, a bellows section, and an integral liner to protect the bellows section. The bellows permit the degree of movement required by the system.

TURBINE EXHAUST DUCT

The turbine exhaust duct is a curved, stainless-steel assembly consisting of mating flange, forward support bracket, bellows section, curved duct, and aft support bracket. The bellows section permits the degree of movement required.

HEAT EXCHANGER

The heat exchanger is a welded, stainless-steel assembly consisting of an outer shell, an inlet flange, an outlet flange, a helix-wound four-coil system, and coil inlet and outlet manifolds. Turbine exhaust gases passing through the shell heat the coils. Liquid oxygen, under turbopump pressure, enters three coils of the four-coil system to be converted to gaseous oxygen for pressurizing oxygen tanks.

ASPIRATOR

The exhaust gas aspirator is a welded, Hastelloy C shell assembly installed over, and extending beyond, the thrust chamber exit on all outboard engines. The aspirator prevents recirculation of fuel-rich gas generator exhaust gases into the missile boattail during flight, by directing these gases into the exit flow stream. The aspirator is welded to the thrust chamber forward channel and located approximately 20 inches forward of the thrust chamber exit.

Gas Generator and Control System

The gas generator and control system consists of the liquid propellant gas generator, ignition monitor valve, purge check valve, orifices, bootstrap lines, and the hose and line assemblies which make up the series control line. The gas generator and control system control engine start sequencing and supply power to drive the turbopump.

LIQUID PROPELLANT GAS GENERATOR

The liquid propellant gas generator (GG) produces combustion gases, during mainstage operation, to drive the two-stage turbine, which in turn supplies power through a reduction gear train to drive the propellant pumps.

The assembly consists of the gas generator control valve, injector assembly, and combustion chamber. The control valve oxidizer poppet and fuel poppet are opened by pressure sensed at the thrust chamber fuel injector manifold when main propellant ignition is established. The propellants flow through the open poppets, through separate manifolds and orifices in the injector to the combustion chamber. The propellants are ignited by the solid propellant gas generator hot gases augmented by the two auto-ignition igniters. The gas generator injector is a uniform mixture-ratio type featuring two fuel streams impinging on a single oxidizer stream.

GAS GENERATOR CONTROL VALVE

The control valve is a normally closed valve with linked poppets to control the flow of propellants to the gas generator injector and combustion chamber. The valve contains a fuel poppet, oxidizer poppet, piston, fuel poppet spring, and oxidizer poppet bellows assembly. These units are contained on a single housing which is bolted to the injector.

The control valve is opened by control (fuel) pressure at main propellant ignition. This pressure, sensed at the thrust chamber fuel injector port, forces the piston on the fuel side of the valve down to open the fuel poppet. A yoke integral with the piston opens the oxidizer poppet. During engine shutdown, the control pressure decays, and the fuel and oxidizer poppets are closed by spring pressure.

The propellant injector cavity design permits an oxidizer lead during start to prevent detonation, and the yoke design ensures a fuel-rich cutoff to eliminate the possibility of turbine burning.

IGNITION MONITOR VALVE

The ignition monitor valve (IMV) is a three-way, pressure-actuated valve that physically senses satisfactory thrust chamber ignition before directing control (fuel) pressure to open the main fuel valve. The valve is mounted on the main fuel valve and is connected to the main fuel valve opening port by a close-coupled orifice fitting.

During engine start, the main LOX valve opens the igniter fuel valve which directs igniter fuel to the hypergol container and fuel pressure to the ignition monitor valve inlet port for subsequent main fuel valve actuation. When satisfactory thrust chamber ignition has been achieved, the pressure buildup sensed at the thrust chamber fuel injector manifold will open the ignition monitor valve. If ignition is not established, ignition monitor valve actuation pressure will not be available; therefore, the main fuel valve will not open.

During engine shutdown, decreasing pressures allow the main fuel valve actuator spring to close the valve and the ignition monitor valve to vent and close.

SOLID PROPELLANT GAS GENERATOR

The solid propellant gas generator (SPGG) is a solid propellant cartridge that bolts to the liquid propellant gas generator flange and supplies power to the turbine for engine starting. It is a disposable unit that cannot be reloaded or reused. The mounting end is closed by a burst diaphragm and an orifice retaining plate. Threaded bosses are provided for two initiators which activate the solid propellant gas generator.

The engine start signal energizes both solid propellant gas generator initiators. As the solid propellant grains start to burn, the burst diaphragm ruptures (pressure is approximately 650 psi)releasing a gas flowrate of approximately 4.68 pounds per second; it will maintain this constant flowrate for approximately 1.0 second. These gases spin the turbine, which in turn drives the LOX and fuel pumps until fuel control pressure opens the liquid propellant gas generator control valve.

The liquid propellant gas generator begins to receive bootstrap propellants from the turbopump propellant discharge ducts as solid propellant grains are consumed. Ignition of liquid propellant gas generator propellants is accomplished by solid propellant grain, which burns approximately 100-200 milliseconds after LOX and fuel enter the combustor. Ignition of liquid propellants is ensured by the use of two auto-ignition igniters.

MAIN LOX VALVE CLOSING VALVE

Closing the main LOX valve and subsequent shutdown of the engine is accomplished by firing a single-body opposed, pyrotechnic-actuated, control valve (CONAX valve). This valve is designed so that actuation of either one or both trigger assemblies will allow fuel pump outlet pressure to flow through the valve body to the closing port of the main LOX valve. Each section of the valve is self-contained and pyrotechnic-actuated. Valve operation is two way and it is normally closed.

Upon receipt of an electrical signal for engine shutdown, the explosive charges within the valve igniter produce the mechanical force required to shear the metal membrane in the valve body, allowing high-pressure control fuel to flow through and on to the closing control port of the main LOX valve. Normal shutdown will result with the operation of only one pyrotechnic actuator.

HYPERGOL CONTAINER

The unitized hypergol container is an integral part of the thrust chamber injector and includes a cylindrical housing to accommodate a six-cubic-inch hypergol cartridge and a hypergol-installed detector switch. A lockpin is provided to secure the cartridge or closure in the container.

GAS GENERATOR IGNITERS AND SOLID PROPELLANT GAS GENERATOR INITIATORS

Igniters and initiators are pyrotechnic devices used to initiate burning of the propellants in the liquid propellant gas generator and the solid propellant gas generator for engine starts.

Igniters

The gas generator auto-ignition igniter is a pyrotechnic device used to ensure ignition of the liquid propellant mixture in the gas generator combustor. Two igniters are required on the gas generator.

Initiators

The solid propellant gas generator initiator is a pyrotechnic device used to initiate burning of the solid propellants in the solid propellant gas generator. Two initiators are used, each consisting of a two-pin electrical receptacle and a moisture sealed cartridge assembly housing a pyrotechnic

material. An electrical impulse of 500 vac, 1.50 amperes minimum, is required to close the circuit through means of a spark gap. Closing of the circuit causes a nichrome wire to turn red hot, thus igniting the pyrotechnic "match-head mix" material. Maximum no-fire voltage is 250 vac.

SYSTEM CHECK VALVES AND COUPLINGS

The check valves on the engine are used to limit the flow of fluids to one direction. Quick-disconnect couplings are used on system fill ports and for system drains.

Propellant Feed System

The propellant feed system consists of the main fuel valve, main oxidizer ignition control valve (consisting of the main LOX valve and the igniter fuel valve), propellant high-pressure ducts, turbopump, check valves, and orifices. The purpose of the propellant feed system is to supply the propellants to the thrust chamber and gas generator.

MAIN LOX VALVE

The main LOX valve (MLV), which is used for controlling the flow of oxidizer to the thrust chamber, is installed between the thrust chamber inlet elbow and the high-pressure duct. The actuation cylinder on the main LOX valve is equipped with a blanket-type heater to prevent abnormal valve operation due to the low temperature of the oxidizer. An acceptable temperature range is 70° to 130°F.

The valve actuation pressure, supplied by the fuel pump discharge, enters the opening port and applies force on the actuation piston. This pressure overcomes the spring pressure and moves the piston toward the open position. As the piston moves to the valve open position, the piston rod linked to a crank rotates the valve shaft and gate 90 degrees to the open position.

The main LOX valve is closed by firing a pyrotechnic-actuated main LOX valve closing control valve which permits fuel pressure to flow to the closing port of the main LOX valve. This equalizes fuel pressure on both sides of the actuation piston, allowing spring force and the difference in area ratios to actuate the valve to the closed position.

IGNITER FUEL VALVE

The igniter fuel valve (IFV) is attached to, and operated by, the main LOX valve. During engine start, the igniter fuel valve is used to sequence the flow of ignition fuel. The igniter fuel valve is actuated by a cam located on the main LOX valve gate shaft. During engine start; the igniter fuel valve leaves the closed position when the main LOX valve reaches 60 degrees of gate rotation and reaches the full-open position when main LOX valve is open 80 degrees.

The igniter fuel valve remains open, permitting fuel flow to the hypergol container and to the inlet port of the ignition monitor valve for main fuel valve actuation. During engine shutdown, as the main LOX valve starts to close, the igniter fuel valve closes, shutting off fuel flow to the thrust chamber igniter fuel spray disks and to the ignition monitor valve. The main fuel valve then closes by spring action and vents into the igniter fuel system.

MAIN FUEL VALVE

The main fuel valve (MFV) is used to control the flow of fuel to the engine and is located between the thrust chamber fuel inlet manifold and the high-pressure fuel duct. The main fuel valve is a butterfly-type with a balanced gate 4.25 inches in diameter and is nearly identical to the main LOX valve except for the difference in the closing control and the absence of the heater assembly and the cam for igniter fuel valve actuation.

When satisfactory ignition has been achieved, the pressure buildup sensed at the thrust chamber fuel injector manifold will open the ignition monitor valve which directs fuel actuation pressure to the opening port of the main fuel valve for valve actuation.

During engine shutdown, main fuel valve closing is accomplished by decreasing pressure allowing the main fuel valve to start closing and the ignition monitor valve to vent actuation pressure into the igniter fuel system. The main fuel valve closes under spring tension.

Turbopump

The turbopump is a turbine driven, dual-pumping unit consisting of an oxidizer pump, a fuel pump, a reduction gearbox, an accessory drive adapter, and a turbine. To simplify the engine system high-pressure plumbing, the turbopump is mounted on the side of the thrust chamber with the main shaft at right angles to the thrust vector. This mounting provides a high-pressure duct routing with minimum pressure drop, reducing the requirements for development of high pump-outlet pressures. The outlets of the oxidizer pump and the fuel pump are integral parts of the respective pump volutes. These outlets are attached to the main propellant ducting.

During engine operation, the turbopump supplies oxidizer and fuel to the thrust chamber at the required pressures and flowrates. The turbopump also supplies the liquid propellant gas generator with the required flow of oxidizer and fuel.

Design refinements incorporated in the latest configuration turbopump include increased volute strength, integral diffuser vanes, high-strength bolting arrangement, and tapered inducer vanes. The gearbox has increased rigidity through use of additional dowels and bolts in the area of the turbine mount. Wider gear teeth and higher purity of material of the intermediate gear and high speed pinion have been incorporated.

Integral accessory drive pads, internal lube passages for all bearings, and adjustable lube pressure by means of a fixed orifice are additional changes made to the gearbox. Turbine seal design is improved; and an electronic tachometer, which uses a single element magnetic pickup to sense turbine shaft speed by means of holes in the inner bearing spacers, is incorporated.

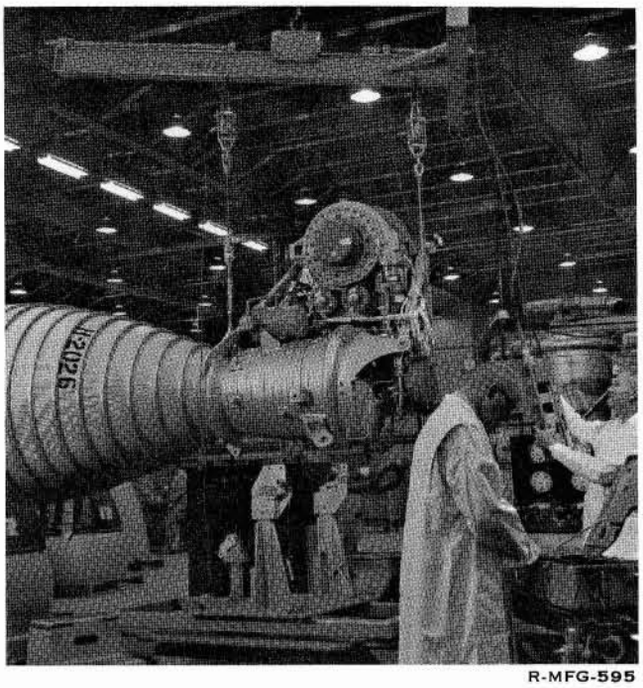
OXIDIZER AND FUEL PUMPS

The turbopump incorporates two single-entry, centrifugal propellant pumps mounted back to back, one on each side of the gearbox. The fuel pump is bolted to the gearbox, and the oxidizer pump is secured to the gearbox by radially inserted steel pins. The steel pins allow the oxidizer pump housing to expand and contract during the extreme temperature changes without distortion and misalignment. Both pumps are driven by a common shaft, and each pump has an axial-flow inducer, a radial-flow impeller, and diffuser vanes.

The oxidizer pump and the fuel pump pressurize the propellants for thrust chamber and gas generator combustion. The axial-flow inducers increase the pressure at the impeller inlet and allow a lower net positive suction head. Hollow-vaned, radial-flow pump impellers are used for pumping the propellants. The propellants pass from the inducers into the impeller inlets, through the impeller slinger vanes to stationary diffuser vanes on the pump adapter and into the pump volutes. The diffuser vanes give uniform distribution of pressure and reduction of fluid velocity around the impellers.

GEARCASE

The turbopump gearcase includes a series of full-depth reduction gears with integral bearing inner races; gear carrier and main shaft bearings; accessory drives; pump shaft bearing seals; and on the oxidizer pump shaft, a gearing heater. The drain manifold is designed for horizontal drainage and includes internal passages for fuel additive lubrication. The turbopump gearcase reduces the speed between the turbine and the pump shaft.



R-MFG-595

Final Assembly — H-1 engine is moved onto final assembly line at Neosho, Missouri plant of Rocketdyne, a division of North American Aviation, Inc. The H-1 engine, which develops 200,000 pounds of thrust, is being produced for Marshall Space Flight Center. It is used in a cluster of eight, developing 1.6 million pounds of thrust, in the S-IB stage.

TURBINE

The turbopump turbine is an impluse-type, two-stage, pressure-compounded unit used to drive the turbopump. The turbine is bolted to the fuel pump housing and consists of an inlet manifold, first- and second-stage turbine wheels and nozzles, a turbine shaft, and a splined quill shaft connecting the turbine shaft to a high-speed pinion gear.

A calibrated, liquid propellant gas generator system controls the flow of hot gases which drive the turbine. The turbine inlet manifold distributes the gases to the first-stage turbine wheel. After passing through the first-stage turbine wheel, the gases increase in velocity by passing through the second nozzle and the second-stage turbine wheel. The gases then leave the turbine through the exhaust ducting. A seal, installed between the first- and second-stage wheels, prevents the hot gas from bypassing the second-stage nozzle. Two-stage sealing also prevents high-temperature gases from heating the turbine shaft bearings.

Fuel Additive Blender Unit

A fuel additive blender unit is incorporated into the engine system to reduce weight by eliminating the oil tank, pressurizing equipment, plumbing, and controls which were required by previous lubrication systems. Extreme-pressure additive RBO140-006 (Rocketdyne) is injected into, and mixed with, RP-1 fuel from the fuel pump discharge. This mixture is introduced into the turbopump gearbox where it lubricates and cools the gearbox components.

ELECTRICAL SYSTEM

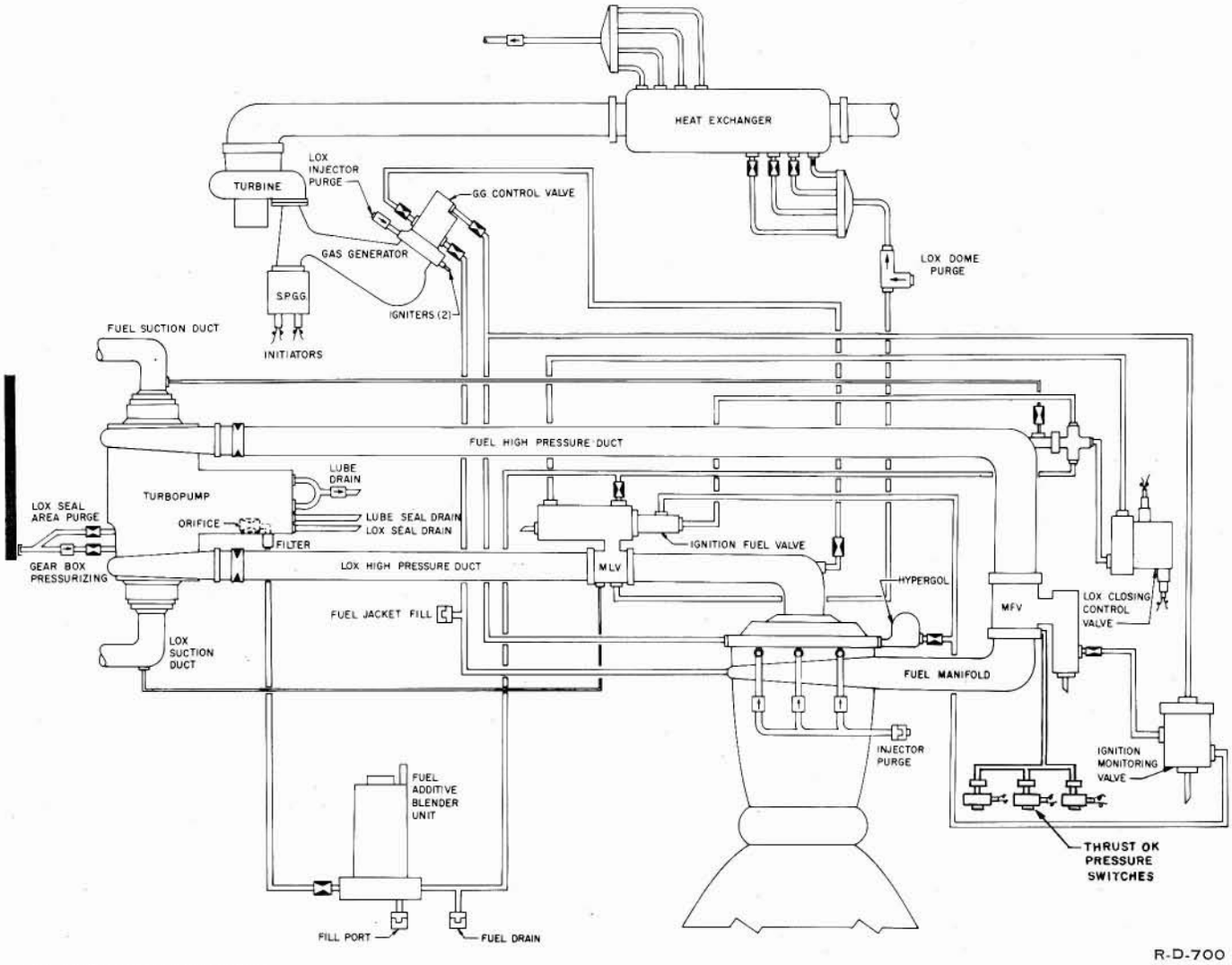
The engine electrical system consists of armored and unarmored electrical harnesses, switches, component heaters, and thrust-OK pressure switches. The thrust-OK pressure switch will commit launch automatically and signal that the engine has attained mainstage operation. In flight this system will provide a cutoff signal in the event thrust falls below a predetermined operating level. The electrical system will condition and check out the engine and control engine start, flight, and cutoff.

ENGINE OPERATION

Ignition Stage

When engine prefiring preparations are complete, the engine firing switch is manually actuated, sending an electrical signal to the solid propellant gas generator initiators, which ignite the solid propellant charge. Burning of the solid propellant accelerates the turbine, causing turbopump discharge pressures to increase against the closed propellant valves. Fuel pressure tapped off the fuel high-pressure duct by the series control line is directed to the igniter fuel valve inlet, fuel additive blender unit inlet, opening port of the main LOX valve, and the closing control manifold.

When fuel pump discharge pressure attains 70 to 110 psig, the fuel additive blender unit opens to



H-1 Engine Schematic Diagram

permit lube flow to the turbopump bearings and gears. When fuel pump discharge pressure reaches 300 ± 50 psig, the main LOX valve opens and the LOX dome, injector, LOX bootstrap, and heat exchanger supply lines start to prime.

The igniter fuel valve is opened by a cam on the main LOX valve gate shaft when the main LOX valve is opened. This directs fuel pressure to the hypergol container inlet and to the inlet port of the ignition monitor valve.

When fuel pressure at the hypergol inlet reaches approximately 300 psig, the hypergol cartridge burst diaphragms rupture and pyrophoric fluid, followed by ignition fuel flows through the thrust chamber injector igniter fuel spray housings and into the combustion area where it contacts the LOX, ignites, and burns, completing the ignition phase of engine starting.

Transition

If a satisfactory thrust chamber ignition has been achieved, a pressure increase in the thrust chamber fuel injector manifold will be sensed at the ignition monitor valve control port. Sufficient pressure at the control port will open the ignition monitor valve which allows fuel control pressure to open the main fuel valve.

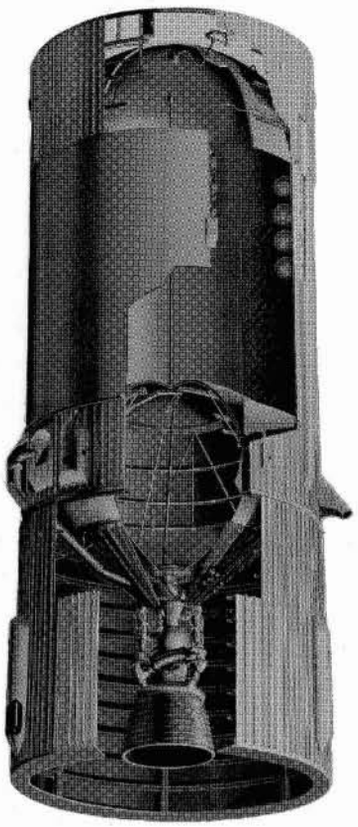
Fuel, under turbopump pressure, flows through the open main fuel valve, to the fuel bootstrap line, through the thrust chamber fuel jacket, through the injector, and into the thrust chamber combustion zone where main propellant ignition is accomplished.

Increasing fuel pressure, resulting from main propellant ignition and sensed at the thrust chamber fuel injector manifold, opens the liquid propellant gas generator control valve. Bootstrap propellants enter the liquid propellant gas generator with a slight LOX lead and are ignited by hot gases from the solid propellant gas generator and the two auto-ignition igniters. The turbopump accelerates, the thrust builds up, and rated operation is attained.

Engine Cutoff

Engine shutdown is achieved by an electrical cutoff signal which fires a pyrotechnic-actuated valve to close the main LOX valve, cutting off LOX flow to the thrust chamber and gas generator. Closing the main LOX valve allows the igniter fuel valve to close, shutting off fuel pressure to the hypergol container inlet, ignition monitor valve inlet, and the main valve opening port. When the main fuel valve actuation pressure decays to approximately 200 psig, the main fuel valve closes under spring pressure, completing engine shutdown approximately 1 second after cutoff signal. A fuel-rich shutdown is provided to prevent a temperature spike in the liquid propellant gas generator. A natural fuel-rich shutdown occurs in the thrust chamber.

S-IVB STAGE FACT SHEET



D-PB-202

DIAMETER: 21.7 ft
HEIGHT: 58.4 ft
WEIGHT: 23,000 lb (empty)
253,000 lb (loaded)
BURN TIME: 470 sec (approx)

MAJOR STRUCTURAL COMPONENTS

AFT INTERSTAGE
AFT SKIRT
THRUST STRUCTURE
PROPELLANT TANK
COMMON BULKHEAD
FORWARD SKIRT

MAJOR SYSTEMS

PROPULSION: One bipropellant J-2 engine
Total Thrust: 200,000 lb
Propellants: Liquid hydrogen 38,000 lb (64,000 gal)
Liquid oxygen 191,000 lb (20,000 gal)
HYDRAULIC: Power for gimbaling J-2 engine.
ELECTRICAL: One 56 vdc and three 28 vdc batteries provide basic power for all electrical functions.
TELEMETRY AND INSTRUMENTATION: Five modulation subsystems provide transmission of flight data to ground stations.
ENVIRONMENTAL CONTROL: Provides temperature controlled environment for components in aft skirt and aft interstage.
ORDNANCE: Provides explosive power for stage separation, retrorocket ignition, ullage rocket ignition and jettison, and range safety requirements.
FLIGHT CONTROL: Provides stage attitude control and propellant ullage control.

S-IVB STAGE

STAGE DESCRIPTION

Basically, the S-IVB stage is an aluminum airframe structure powered by a single 200,000 pound-thrust J-2 engine which burns liquid oxygen and liquid hydrogen. The structure has a bipropellant tank capacity of approximately 230,000 pounds of usable fuel and oxidizer.

STAGE FABRICATION AND ASSEMBLY

The S-IVB stage structure consists of an aft interstage assembly, aft skirt assembly, thrust structure assembly, propellant tank assembly, and forward skirt assembly. The propellant tank assembly consists of a single tank separated by a common bulkhead into a fuel compartment and an oxidizer compartment.

Aft Interstage Assembly

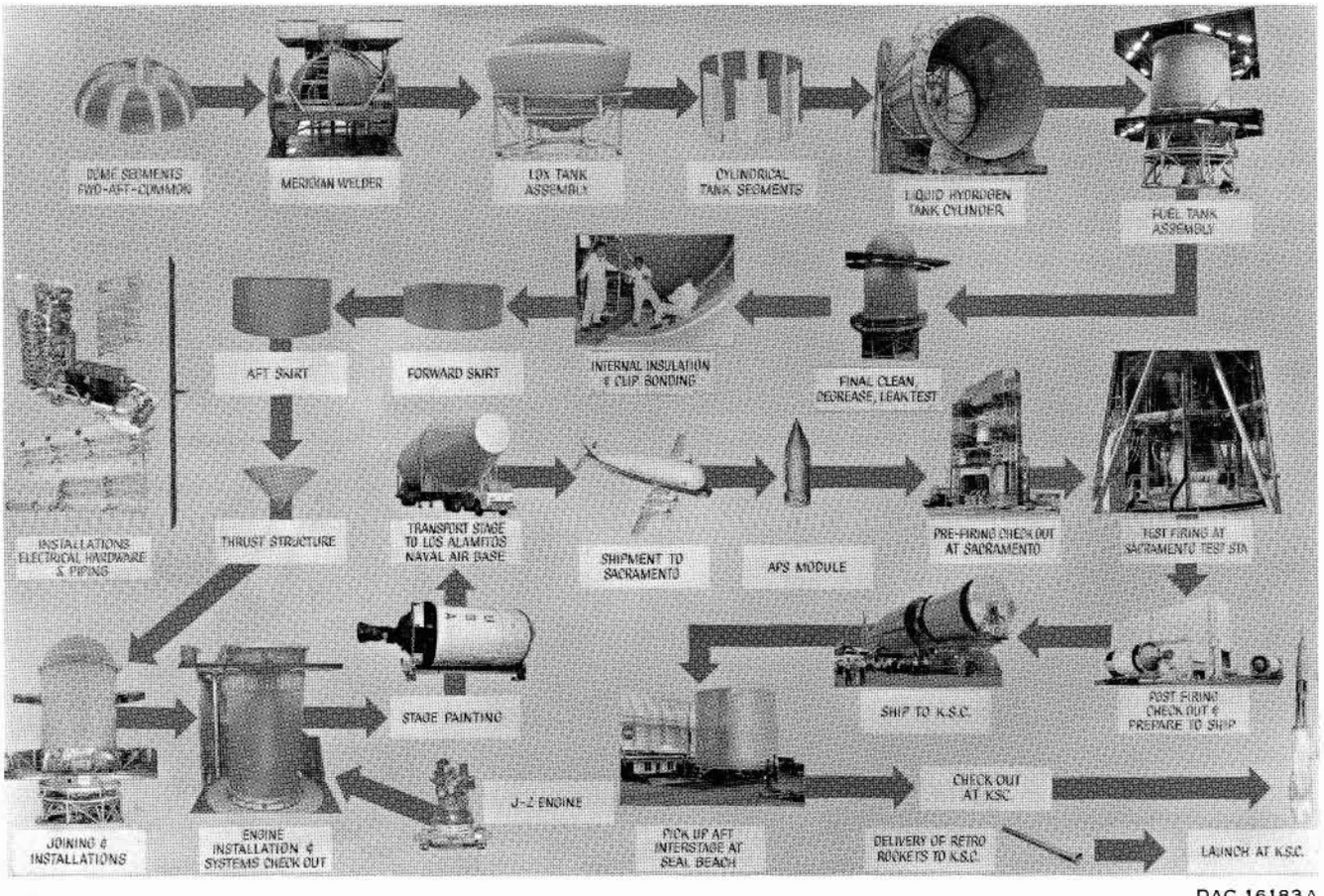
The aft interstage is a cylindrical structure fabricated of stringer-stiffened aluminum skin panels. The aft interstage supports the S-IVB stage and provides a structural interface with the S-IB first stage. Vehicle design is such that the aft interstage remains with the S-IB stage when the two stages are separated in flight. Four solid propellant retrorocket motors are located equidistant around the base of this structure to brake the S-IB stage during separation.

Aft Skirt Assembly

The aft skirt is a cylindrical structure fabricated of aluminum, stringer-stiffened, skin panels and provides structural interface between the aft interstage and propellant tank assembly. After first stage burnout, the S-IB/S-IVB separation plane forms a part of the aft skirt structure. Two auxiliary propulsion system engine modules, three ullage rocket modules, and the aft umbilical connector plate are located on the aft skirt assembly.

Thrust Structure Assembly

The thrust structure is a truncated, cone-shaped structure fabricated of aluminum skin and stringers. It is bolted to the aft dome of the propellant tank assembly to provide an attach point for the J-2 engine. The thrust structure distributes J-2 engine thrust over the entire tank circumference. In addition, hydraulic system components, propellant feed lines, pneumatic components, and miscellaneous components which support engine operation are mounted on the thrust structure assembly.


DOME SEGMENTS FWD-AFT-COMMON
MERIDIAN WELDER
LOX TANK ASSEMBLY
CYLINDRICAL TANK SEGMENTS
LIQUID HYDROGEN TANK CYLINDER
FUEL TANK ASSEMBLY
AFT SKIRT
FORWARD SKIRT
INTERNAL INSULATION & CLIP BONDING
FINAL CLEAN, DEGREASE, LEAK TEST
INSTALLATIONS ELECTRICAL HARDWARE & PIPING
THRUST STRUCTURE
TRANSPORT STAGE TO LOS ALAMITOS NAVAL AIR BASE
SHIPMENT TO SACRAMENTO
APS MODULE
PRE-FIRING CHECK OUT AT SACRAMENTO
TEST FIRING AT SACRAMENTO TEST STA
STAGE PAINTING
SHIP TO K.S.C.
POST FIRING CHECK OUT & PREPARE TO SHIP
J-2 ENGINE
CHECK OUT AT KSC
JOINING & INSTALLATIONS
ENGINE INSTALLATION & SYSTEMS CHECK OUT
PICK UP AFT INTERSTAGE AT SEAL BEACH
DELIVERY OF RETRO ROCKETS TO K.S.C.
LAUNCH AT K.S.C.

DAC 16183A

S-IVB Stage Production Sequence

Propellant Tank Assembly

Structural elements of the propellant tank assembly are a cylindrical tank section, common bulkhead, aft dome, and forward dome. Seven segments are machined from aluminum alloy plate to form the tank section. A waffle pattern is then machine-milled into each segment to reduce weight and provide shell stiffness. The formed segments are joined into a complete cylinder by single-pass internal weld on a Pandjiris welding machine.

Aft and forward domes are made by forming "orange peel" segments on a stretch press. Orange peel segments are then joined in a dome welder. Each dome assembly rotates in the fixture under a stationary welding head which is automatically positioned by a servo-controlled sensing element. To complete the hemisphere, a 30-inch "dollar" segment is welded in the top center of the dome. Subsequently, all fittings for various tank connections are installed by machine weld.

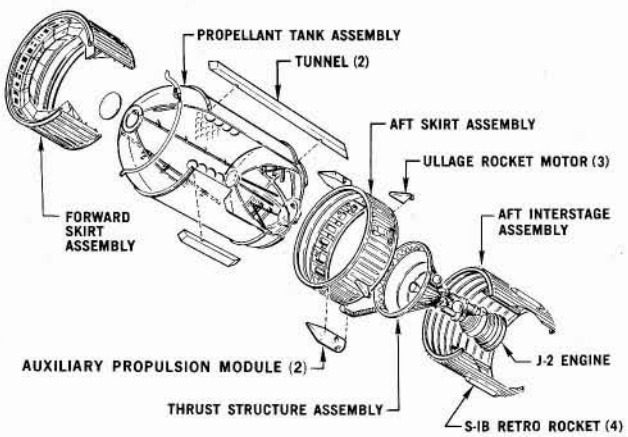
COMMON BULKHEAD

The common bulkhead, which forms the physical separation between the LOX and hydrogen tanks, is a 130-inch-radius hemisphere consisting of two aluminum domes separated and insulated by a fiberglass honeycomb core. The honeycomb core is bonded between the two domes under heat and pressure. Edges of two peripheral tees are welded together to provide a seal for the core, and structurally join the top and bottom domes.

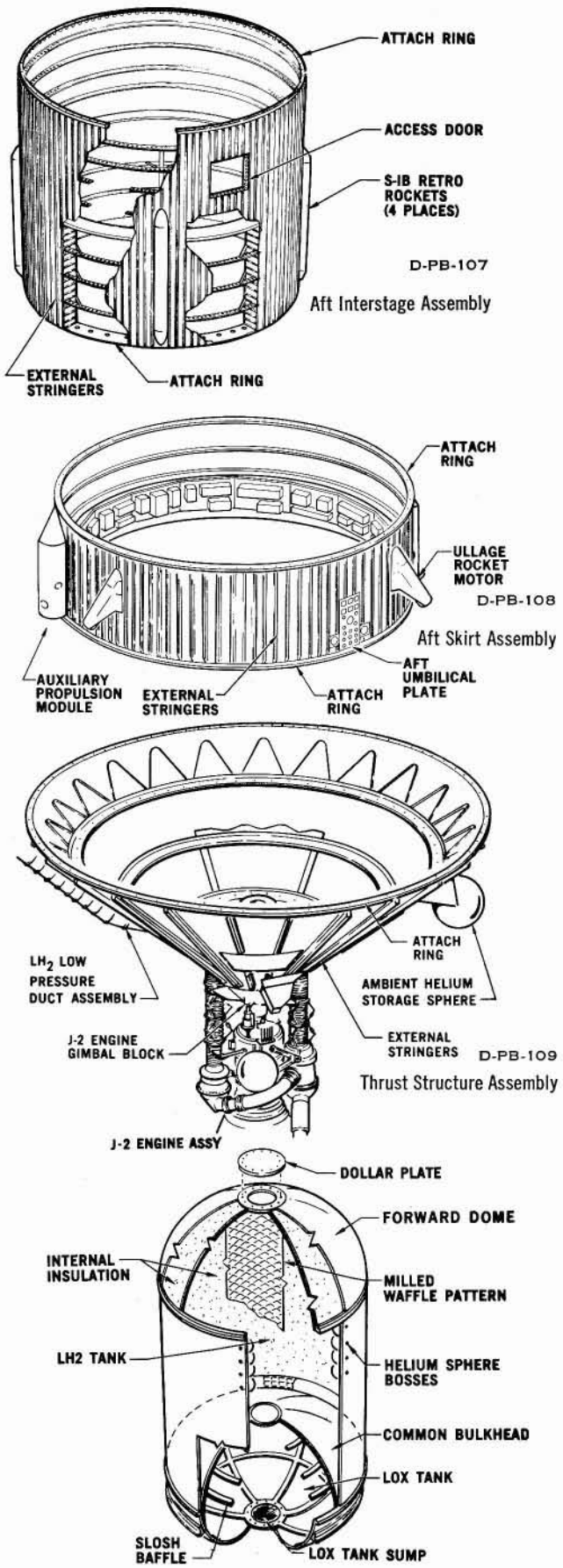
Joining the common bulkhead and the aft dome completes the LOX tank subassembly. A slosh-baffle located within the LOX tank breaks up any wave action of the oxidizer during flight. The baffle is made up of four rings supported by the "A" frames.

Forward Skirt Assembly

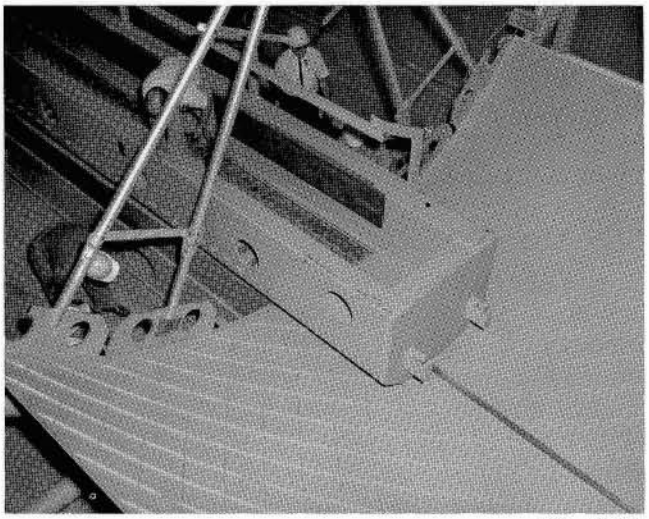
The forward skirt is a cylindrical aluminum skin and stringer structure that provides a hard attach



S-IVB Stage Exploded View



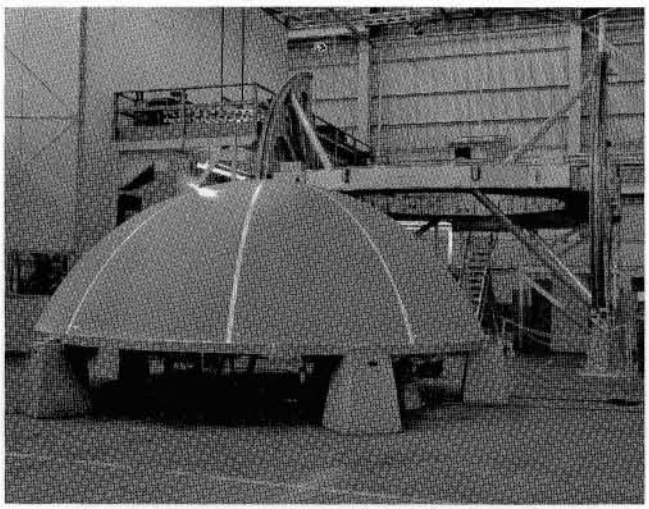
Propellant Tank Assembly



DAC 11963

Longitudinal Weld — The automatic welding fixture performing a longitudinal weld on LH_2 tank segment.

point for the IU. In addition, the forward skirt provides an interior mounting structure for electrical and electronic equipment that requires environmental conditioning, as well as range safety and telemetry antennas mounted around the exterior periphery. Environmental conditioning for electronic equipment is provided by cold plates which utilize a coolant supplied from the IU thermo conditioning system.



D-SSC005321

Meridian Welder — Illustrated is the Meridian welder which produces domes from prefabricated "orange peel" segments.

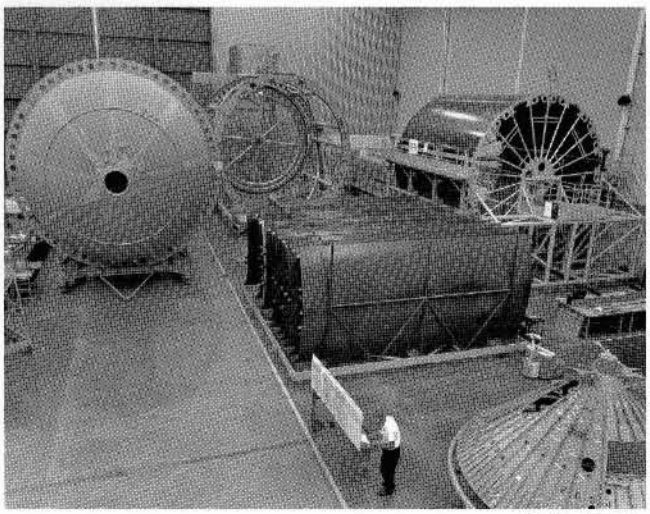
Final Assembly

Final assembly of the S-IVB stage propellant tank structural components is accomplished in the assembly and welding tower. The assembly is then removed from the tower and transported to the insulation chambers building where the LH_2 tank insulating tiles are fitted and installed, a glass cloth liner placed on the insulation and a sealant added. Propulsion system components, internally mounted in the LH_2 tank, are installed following the completion of tank insulating. The structure is then returned to the assembly tower where the foward and aft skirts and thrust structure are installed.

Final installation of various subsystem components is performed in a checkout tower, along with the installation and alignment of the J-2 engine. The stage is in a vertical position in the tower

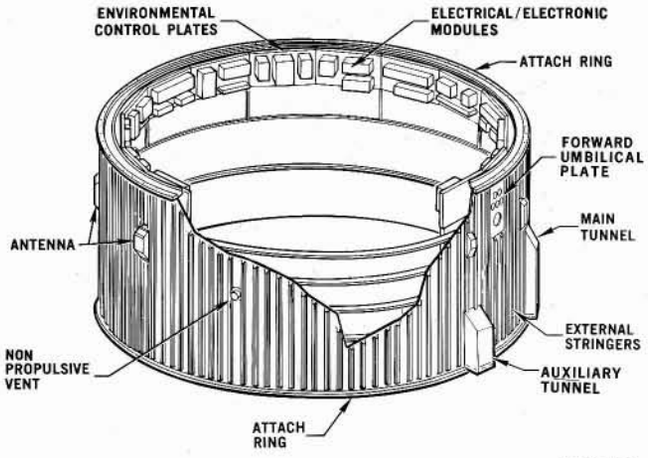
D-SM400327

Common Bulkhead — Shell is lowered onto the lower dome for fit check before honeycomb installation.



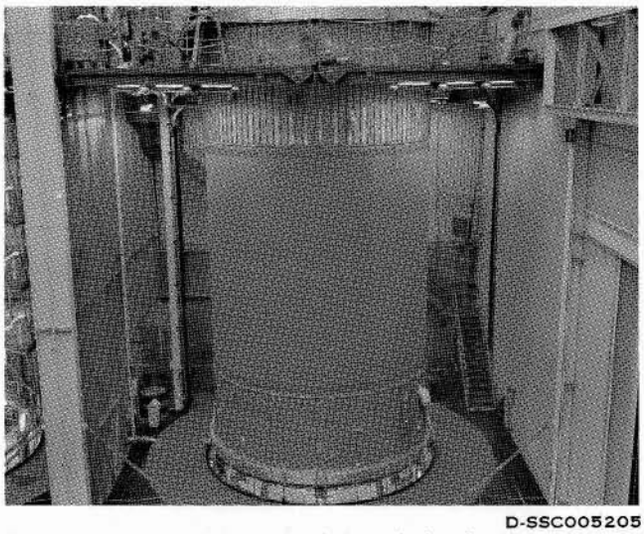
DAC 17793

Propellant Tank Assembly Area — The two large fixtures are used for tank segment welding and trimming of attach angles.



D-PB-111

Forward Skirt Assembly



D-SSC005205

Tower No. 2 — Forward and aft skirts and the thrust structure are installed in this manufacturing position.

DAC 10274

Final Installation of Tank Tiles — Location numbers are inscribed on each tile.

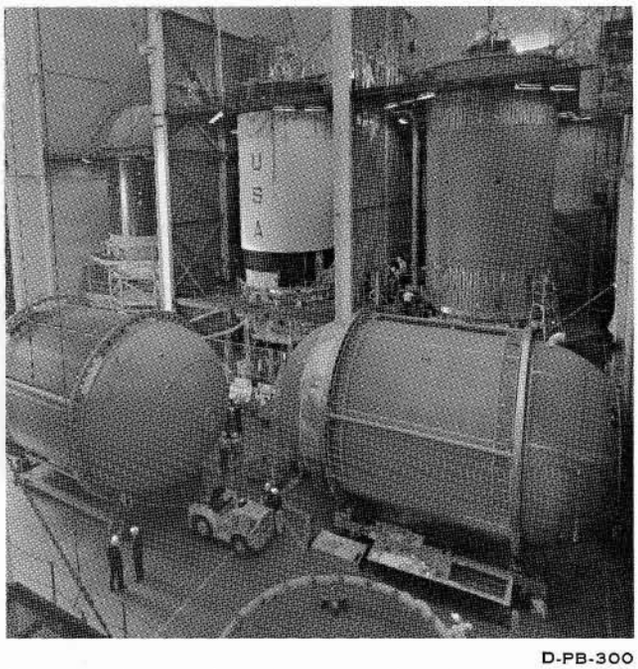
where a complete stage checkout is conducted. After satisfactory checkout, the stage is removed from the tower, placed on a dolly, and ground support rings are installed at each end of the stage. It is then painted, weighed, and prepared for shipment to the Douglas Sacramento Test Center.

S-IVB STAGE SYSTEMS

Major systems required for S-IVB stage operation are the propulsion system, flight control system, electrical power and distribution system, instrumentation and telemetry system, environmental control system, and ordnance system.

Propulsion System

The propulsion system consists of the J-2 engine, propellant system, pneumatic control system and propellant utilization system. The J-2 engine burns LOX as an oxidizer and LH_2 as fuel at a nominal mixture ratio of 5:1. Both fuel and oxidizer systems have a vent and relief capability to protect the propellant tanks from overpressurization. The pneumatic control system regulates and controls both the oxidizer and fuel systems. The propellant utilization (PU) system assures simultaneous and precise fuel and oxidizer depletion by controlling engine mixture ratio.



D-PB-300

Space Age Assembly Line — Rapid pace of hardware production for America's lunar exploration program is shown in this photo of five S-IVB stages in various phases of assembly at the Douglas Space Systems Center, Huntington Beach, Calif. In vertical towers at right and center, two stages undergo final checkout before shipment, while another is in position for cleaning and degreasing operations at left. On floor at right, work is being done inside a stage; and at left, another stage awaits hoisting into tower.

J-2 ENGINE

The engine system consists of the J-2 engine, propellant feed system, start system, gas generator system, control system, and a flight instrumentation system. The propellant feed system utilizes independently driven, direct drive fuel and oxidizer turbopumps to supply propellants at the proper mixture ratio to the engine combustion chamber. Additional information on the J-2 engine system may be found in the J-2 Engine section.

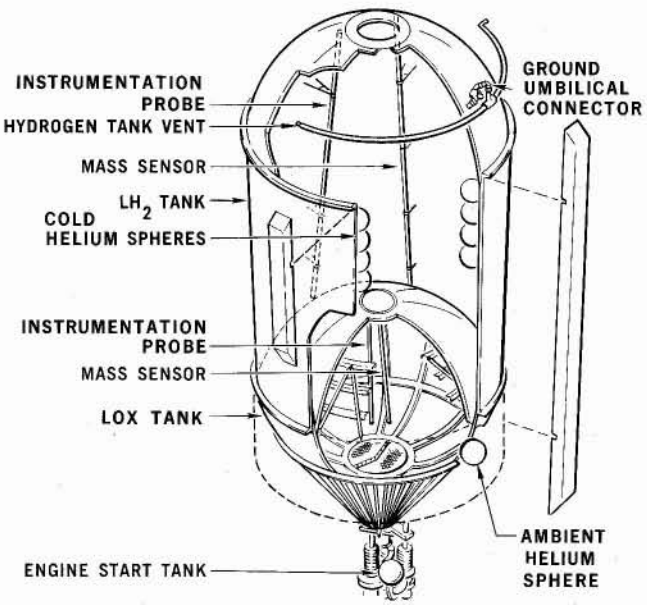
PROPELLANT SYSTEM

The propellant system consists of related stage subsystems to support propulsion of the S-IVB, and includes the oxidizer system, fuel system, pressurization system, tank venting system, and chilldown recirculation system.

Oxidizer System

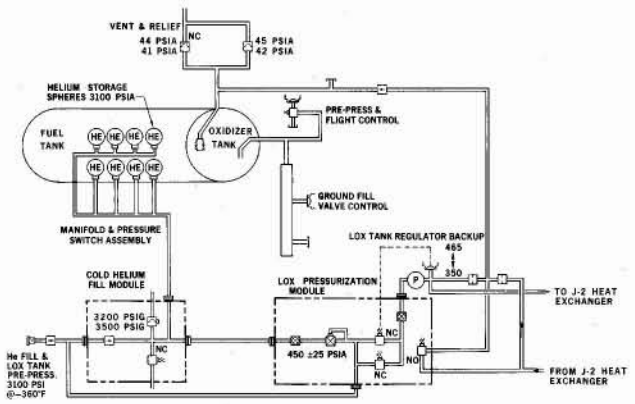
LOX is loaded into the LOX tank at a temperature of —297°F. The LOX tank capacity is 2,828 cubic feet which provides tankage for approximately 191,000 pounds (20,000 gallons) of usable LOX. The tank is pressurized with gaseous helium at 37 to 40 psia, and is maintained at this pressure during liftoff, boost, and stage engine operation.

Fill and Drain — The LOX filling operation consists of purging and chilldown of the tank, and filling in four stages: slow fill, fast fill, replenish (topping), and pressurization. The ground controlled combination vent and relief valve is pneumatically opened at the start of the fill operation. During slow fill LOX is loaded at a rate of 300 gpm until a 5 per cent of full level is attained, then fast fill at 1,000 gpm is initiated. When 93 per cent of



D-PB-112

Propulsion System Components



D-PB-114

Oxidizer Tank Pressurization System

the LOX has been loaded, the fill rate is reduced to a rate of 0 to 270 gpm. If for any reason, the LOX tank becomes overpressurized during fill, a vent malfunction occurs, or if there is an excessive LOX fill flow, a pressure switch signals the LOX ground fill valve to close.

The LOX tank is capable of being unloaded by reversing the flow through the fill system under tank pressure and/or from gravity effect. Drain capacity is at 500 gpm at 33 psia.

LOX Tank Pressurization — The LOX tank is pressurized at 39.5 ±0.5 psia from a ground supply of cold helium regulated to —360°F. Following liftoff, the LOX tank pressure is maintained from helium storage spheres located in the LH_2 fuel tank that have been charged to 3,000 ±100 psi at —360°F. The engine heat exchanger heats and expands a portion of the helium flow before it is fed into the LOX tank. An ullage tank pressure switch controls inflight pressurization by opening or closing the cold helium flow valve as required. In case of pressure switch failure inflight, a pressure switch and plenum chamber act as a backup pressure regulator.

LOX Tank Vent-Relief System — The LOX tank vent-relief system consists of a tee assembly with a pneumatically-operated vent valve and a backup relief valve. Pneumatic operation is provided by the LOX vent actuation module using helium gas from the pneumatic control system. The vent-relief valve is opened during the ground fill operation and closed prior to pressurization. During fill operations, the vent valve is capable of venting all LOX vapor.

The relief valve backup system automatically relieves at 45 psia and reseats at 42 psia. During liftoff and non-powered stage flight, pressure relief of venting is not anticipated. However, the vent system becomes operational in the event of LOX tank overpressurization.

LOX Feed System — Prior to vehicle liftoff, all LOX feed system components of the J-2 LOX tur-

bopump assembly must be "chilled" to operating temperature for proper operation. Chilldown of the LOX system is accomplished by a closed loop, forward flow recirculation system. On command from the IU, a prevalve in the LOX feed duct closes and a shutoff valve opens. An auxiliary electrically driven centrifugal chilldown pump, mounted in the LOX tank, starts and LOX chilldown circulation begins.

LOX is circulated from the LOX tank, through the low pressure feed duct, to the J-2 engine LOX pump and bleed valve, then back to the LOX tank through return lines. The pump is capable of delivering a minimum flow rate of 31 gpm at 25 psia. Recirculation chilldown continues through the boost phase and up to the time for J-2 engine ignition. In the event of an emergency, the chilldown system shutoff valve closes upon command from the IU.

A low pressure supply duct supplies LOX from the tank to the engine at a nominal flow rate of 391 pounds-per-second at —297° F. at 25 psia and up.

The main LOX feed valve is a 4-inch butterfly type valve and opens in two distinct steps; the first, a partially opened position; the second, a fully opened position. The LOX feed valve is solenoid controlled.

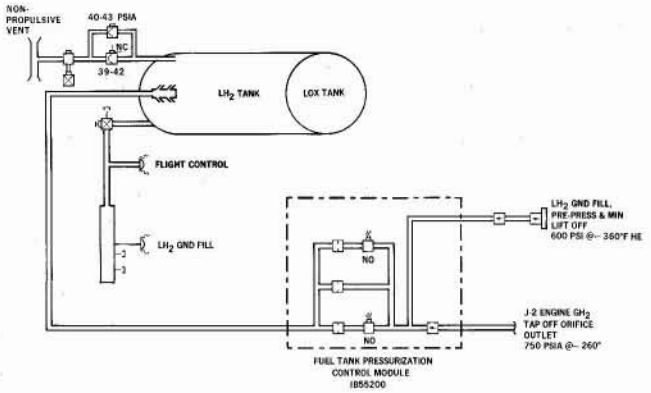
A signal from the engine sequencer energizes the LOX feed valve, as required, to obtain steady-state operation. During steady-state operation, LOX feed is regulated by a propellant utilization valve which controls the oxidizer flow to the engine. A complete description of engine operation may be found in the J-2 Engine section.

FUEL SYSTEM

LH_2 is loaded into the insulated fuel tank at a temperature of —422.9° F. The tank capacity is 10,446 cubic feet, ensuring approximately 38,000 pounds (64,000 gallons) of usable fuel. The tank is pressurized from a ground source of helium at 30.5 ± .5 psia. During liftoff, boost and stage engine operation pressure is maintained in the fuel tank at 28 to 31 psi.

Fill and Drain — The LH_2 loading operation consists of purging, chilldown of the tank, and filling in four stages: slow fill, fast fill, replenish (topping), and pressurization.

Immediately prior to LH_2 input into the tank, a combination vent and relief valve is pneumatically opened. Loading is then initiated into the tank at 500 gpm until 5 per cent of full level is reached, then fast fill begins. During fast fill, LH_2 is supplied to the tank at 3,000 gpm. When 93 per cent of the load is completed, a replenish or topping loading rate is initiated at between 0 to 270 gpm. Filling is automatically terminated at 100 per cent full. During the final topping operation, the fuel tank venting system is closed and the tank is simultaneously pressurized from the ground source of helium.



D-PB-113

Fuel Tank Pressurization System

If overpressurization of the tank should occur during fill, or during the boost phase, a relief valve, which is spring loaded to open at 43 psia and close at 40 psia, is actuated to relieve excess pressure.

The LH_2 tank is capable of being unloaded through the fill system. LH_2 unloading is accomplished by reversing the flow through the fill system under tank pressure and/or from gravity effect.

Fuel Tank Pressurization — During initial tank pressurization, an external tank connection is made to a ground supply of helium. The helium is supplied to the fuel tank at —360° F. at 600 psig. When the tank ullage pressure reaches a maximum of 30.5 ±.5 psia, a pressure switch sends a signal to deactivate the ground pressurization valve indicating that a satisfactory liftoff pressure has been attained, and pressurization is discontinued.

During liftoff and prior to J-2 engine start, additional pressurization is not required, as tank ullage pressure will be maintained from fuel boiloff.

At the initiation of J-2 engine start, GH_2 is bled from the J-2 engine at 750 psia, —260° F. to provide ullage pressure during fuel depletion. The pressure bled from the engine into the fuel tank is controlled by a fuel tank pressurization control module.

Fuel Tank Vent-Relief System — Venting of the LH_2 tank is accomplished by a vent and relief system capable of relieving all excess pressure accumulated from overpressurization or fuel boiloff during fill and flight operation. During fill, vaporization is vented through a self-sealing disconnect located in the forward skirt. During liftoff and flight, the gases are vented overboard through a non-propulsive exhaust.

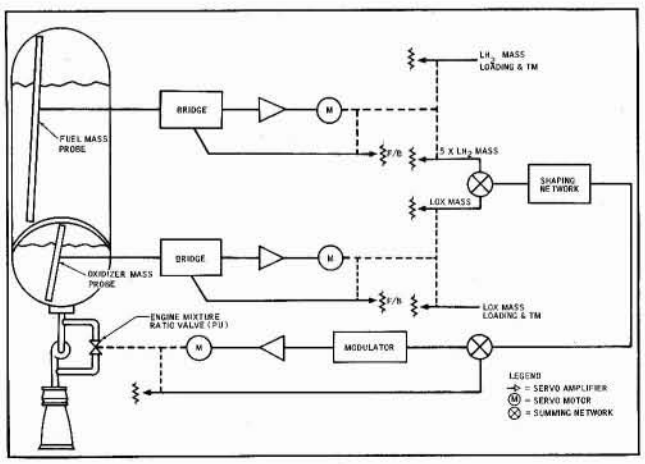
The venting system consists of an actuation control module, vent valve, relief valve, directional control valve, and a non-propulsive overboard exhaust. Actuation of the vent valve is commanded from an external ground signal during fill operations, and from the flight sequencer during liftoff and flight. The vent valve is designed to open in a maximum of 0.1 second upon command.

The relief valve, which provides a backup capability in case of vent valve failure, opens at 42 psia and reseats at 39 psia, and has a flow/relief capability of 2 pounds/second at sea level.

A directional control valve directs excessive pressures through the ground disconnect during fill, and directs excessive pressures through the non-propulsive vent during liftoff and flight. The non-propulsive vent system extends from the directional control valve into two 4-inch vent lines that terminate into two non-propulsive exhaust ports. The ports are located 180° apart, in the forward skirt area. The ports are arranged to direct the exhaust for total thrust cancellation.

LH_2 Feed System—Prior to vehicle liftoff, all LH_2 feed system components of the J-2 turbopump assembly must be "chilled" to assure proper operation. Chilldown of the LH_2 system is accomplished by a closed loop, forward-flow, recirculation system. On command from the IU, the pre-valve in the LH_2 feed duct closes and the chilldown shutoff valve opens. An auxiliary electrically-driven LH_2 chilldown pump mounted in the LH_2 tank, circulates the LH_2 within the system and is capable of a minimum flow rate of 135 gpm at 6.1 psi.

LH_2 is circulated from the LH_2 tank through the low pressure feed duct, through the J-2 engine fuel pump, the fuel bleed valve, and back to the tank through a return line. Recirculation chilldown continues through the boost phase and up to J-2 engine ignition. In the event of an emergency shutdown requirement, the chilldown system shutoff valve is closed upon command from the IU.



D-PB-115

Propellant Utilization System (Simplified)

LH_2 is supplied to the J-2 engine through a vacuum-jacketed, low-pressure duct at a flow rate of 81.26 pounds per second at −423°F., 28 psia. The duct is located in the fuel tank side wall above the common bulkhead joint and is equipped with bellows to compensate for thermal motion. Signals from the engine sequencer energizes the LH_2 feed valve, as required to obtain steady-state operation. A complete description of engine operation may be found in the J-2 Engine section.

PROPELLANT UTILIZATION SYSTEM

The primary function of the PU system is to assure simultaneous depletion of propellants by controlling the LOX flow rate to the J-2 engine. It also provides propellant mass information for controlling the fill and topping valves during propellant loading operations. The system consists of mass sensors, an electronics assembly, and an engine-mounted mixture ratio valve.

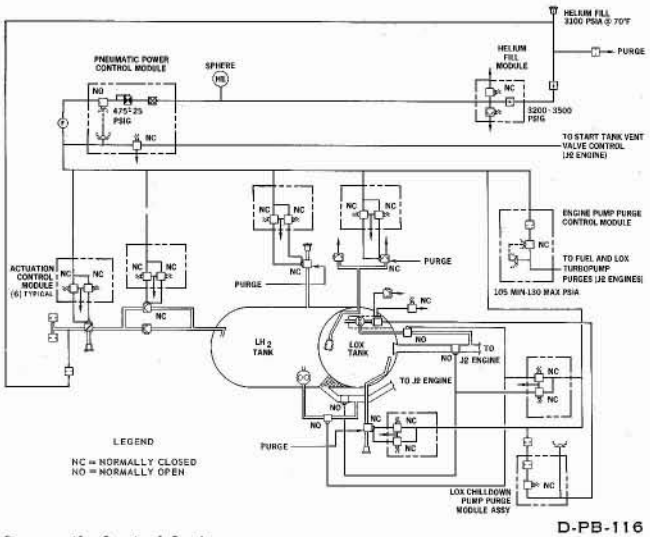
During loading operations, the mass of propellants loaded is determined within 1 per cent by the mass sensors. Tank over-fill sensors act as a backup system in the event the loading system fails to terminate fill operations.

Continuous LH_2 and LOX residual readout signals are provided throughout S-IVB powered flight. The difference between the fuel and oxidizer mass indications, as sensed by the mass sensors, are continually analysed and are then used to control the oxidizer pump bypass flow rate, which changes the engine mixture ratio correspondingly. The static inverter/converter supplies the analog voltages necessary to operate the PU system. It is commanded "on" and "off" by a switch selector and sequencer combination.

PNEUMATIC CONTROL SYSTEM

The pneumatic control system provides GHe pressure to operate all S-IVB stage pneumatically-operated valves with the exception of those provided as components of the J-2 engine. GHe is supplied from an ambient helium sphere, pressurized from a ground source before propellant fill operations, at 3,000 ±100 psia at 70°F. for valve operation. The sphere is located on the thrust structure and is pre-conditioned to above 70°F from the environmental control system before liftoff.

The pneumatic control system provides regulated pressure at 475 ± 25 psig for operation of the LH_2 and LOX vent-relief valves during propellant loading, LH_2 directional control valve, LOX and LH_2



D-PB-116

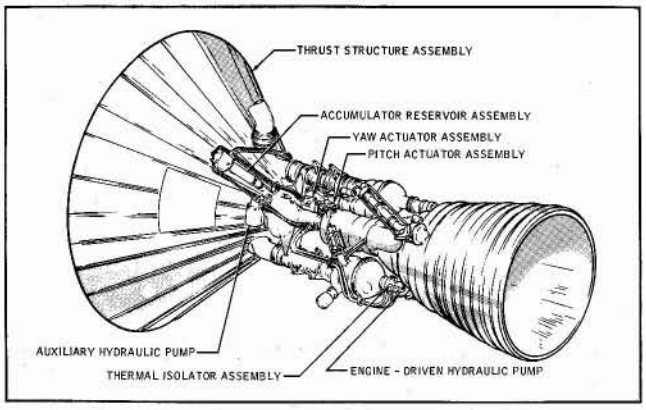
Pneumatic Control System

fill and drain valves during loading, and the GH_2 engine start system vent-relief valve. It also provides operating pressures for the LH_2 and LOX turbopump turbine purge module, LOX chilldown pump purge module control, LOX and LH_2 prevalves, and the LOX and LH_2 chilldown shutoff valves.

The pneumatic control subsystem is protected from overpressure by a normally open solenoid valve controlled by a downstream pressure-sensing switch. At pressures greater than 535 +15, −10 psia, the pressure switch actuates and closes the valve. At pressures below 450 +15, −10 psia, the pressure switch drops out and the solenoid opens, thus acting as a backup regulator.

Flight Control System

The flight control system provides stage thrust vector steering and attitude control. Steering is achieved by gimbaling the J-2 engine during powered flight. Hydraulic actuator assemblies provide J-2 engine deflection rates proportional to steering signal corrections supplied by the IU.



D-ORM147A

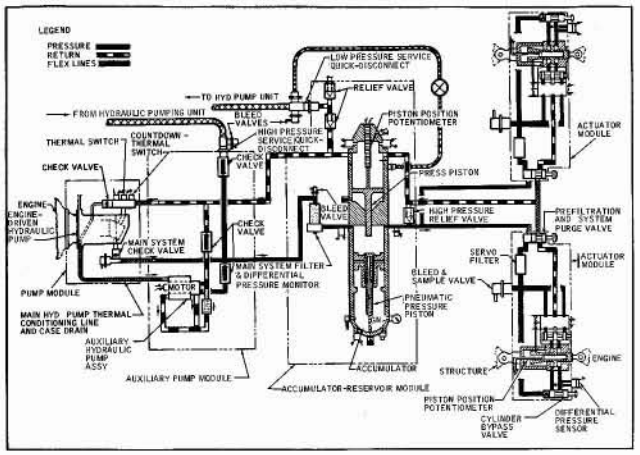
J-2 Engine Hydraulic System Components

Stage roll attitude during powered flight is controlled by firing the auxiliary propulsion system (APS) attitude control engines.

Hydraulic System

The hudraulic system performs engine positioning upon command from the IU. Major components are a J-2 engine-driven hydraulic pump, an electrically driven auxiliary hydraulic pump, two hydraulic accumulator assemblies, and an accumulator-reservoir assembly.

The electrically driven auxiliary hydraulic pump is started before vehicle liftoff to pressurize the hydraulic system. Electric power for the pump is provided by a ground source. At liftoff, the pump is switched to stage battery power. Pressurization of the hydraulic system restrains the J-2 engine



D-ORM148A

Hydraulic Control System (Simplified)

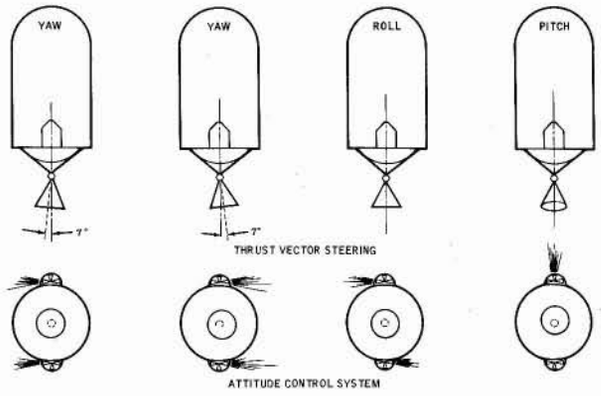
in a null position with relation to the S-IVB stage centerline, preventing pendulum-like shifting from forces encountered during liftoff and boost. During powered flight, the J-2 engine may be gimbaled up to ±7° in a square pattern by the hydraulic system upon command from the IU.

Engine-Driven Hydraulic Pump

The engine-driven hydraulic pump is a variable displacement type pump capable of delivering hydraulic fluid under continuous system pressure and varying volume as required for operation of the hydraulic actuator assemblies. The pump is driven directly from the engine oxidizer turbopump. A thermal isolator in the system controls hydraulic fluid temperature to ensure proper operation.

Auxiliary Hydraulic Pump

The auxiliary hydraulic pump is an electrically driven variable displacement pump which supplies



D-ORM145A

Flight Control System

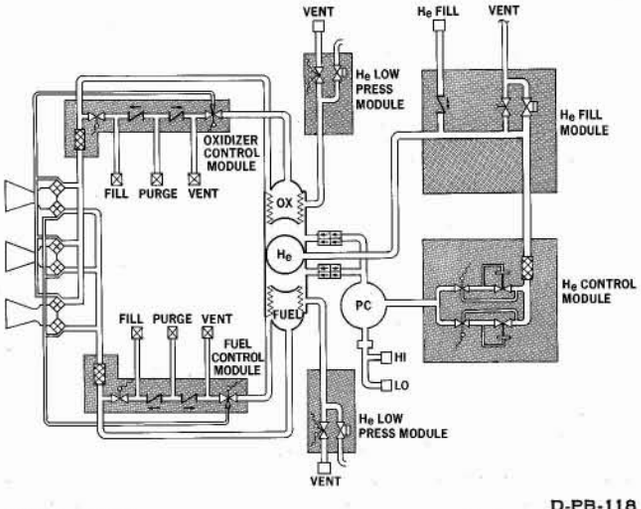
a constant minimum supply of hydraulic fluid to the hydraulic system at all times. The pump is also used to perform preflight engine gimbaling checkouts, hydraulically lock the engine in the null position during boost phase, maintain system hydraulic fluid at operating temperatures during other than the powered phase, and augment the engine-driven hudraulic pump during powered flight. It also provides an emergency backup supply of fluid to the system.

Hydraulic Actuator Assemblies

Two hydraulic actuator assemblies are attached directly to the J-2 engine and the thrust structure, and receive IU command signals to gimbal the engine. The actuator assemblies are identical and interchangeable.

Accumulator-Reservoir Assembly

The accumulator-reservoir assembly is an integral unit mounted on the thrust structure. The reservoir section is the storage area for hydraulic fluid; the accumulator section supplies peak system fluid requirements and dampens high-pressure surges within the system.



D-PB-118

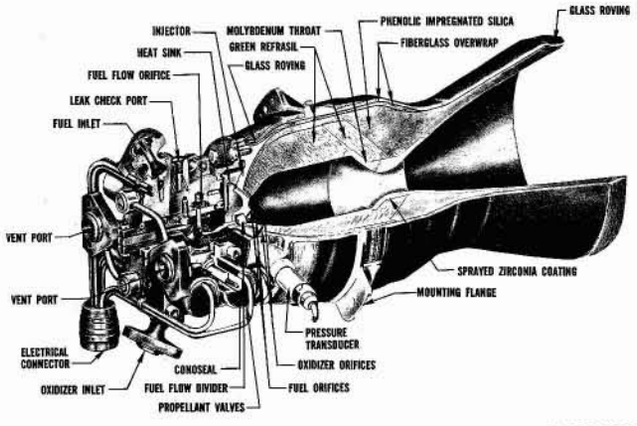
APS Schematic

AUXILIARY PROPULSION SYSTEM

The APS includes modules that provide three-axis attitude control. Two APS modules are mounted 180° apart on the aft skirt assembly. Each APS module contains three, 150-pound-thrust engines. Three solid propellant ullage rocket motors are mounted 120° apart on the aft skirt assembly.

Attitude Control

APS module engines are fired in short bursts for three-axis attitude control during coast. Minimum engine firing pulse duration is approximately 70 milliseconds.



D-PB-119

150-Pound Thrust Attitude Control Engine

Each APS module contains an individual oxidizer system, fuel system, and pressurization system. The modules are self-contained and easily detached for separate checkout and environmental testing.

The individual engines are approximately 15 inches long with exit cones approximately 6.5 inches in diameter. Engine cooling is accomplished by an ablative process.

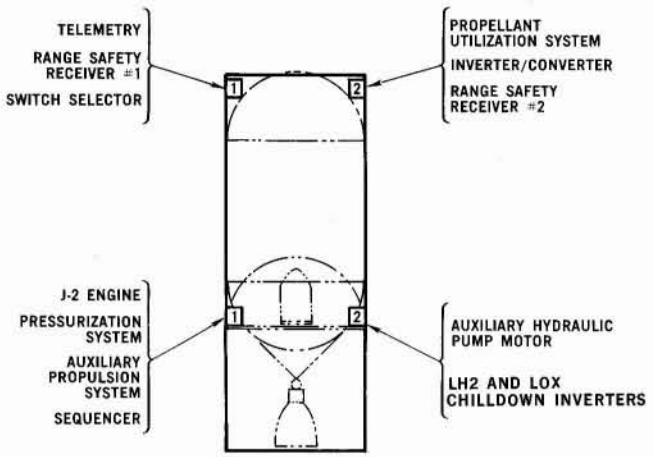
Ignition is unnecessary because fuel and oxidizer are hypergolic (self-igniting). Nitrogen tetroxide (N_2O_4), the oxidizer, is stable at room temperature.

Approximately 37.1 pounds of usable oxidizer is stored in the upper section of a common oxidizer/fuel tank of the expulsion bellows type. Stored high-pressure helium is used for pressurizing oxidizer and fuel tanks.

The fuel, monomethyl hydrazine ($CH_3N_2H_3$), is stable to shock and extreme heat or cold. Approximately 22.9 pounds of usable fuel is stored in the lower section of the common oxidizer/fuel tank.

Ullage Control

Three solid propellant Thiokol TX-280 rocket motors, each rated at 3,390 pounds of thrust, are ignited during separation of the S-IB and S-IVB stages for ullage control. This thrust produces additional positive stage acceleration during separation, and position LOX and LH_2 propellants toward the aft end of their tanks to cover outlets. In addition, propellant boiloff vapors are forced to the forward end where they are safely vented overboard. Tank outlets are covered to insure a net positive suction head (NPSH) to the propellant pumps, thus preventing possible pump cavitation during J-2 engine start.

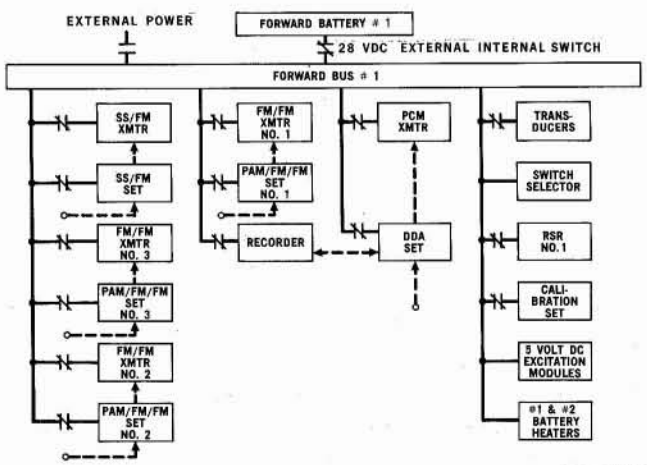


D-PB-120

S-IVB Basic Electrical Power and Distribution System

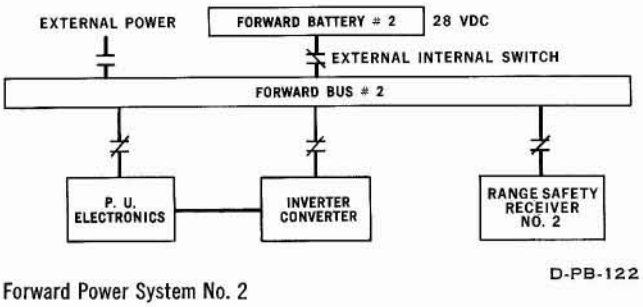
Electrical Power and Distribution System

Four battery-powered systems provide electrical requirements for S-IVB stage operation. Forward Power System No. 1 includes a 28 vdc battery and power distribution equipment for telemetry, range safety system No. 1, forward battery heaters, and a power switch selector located in the forward skirt area.



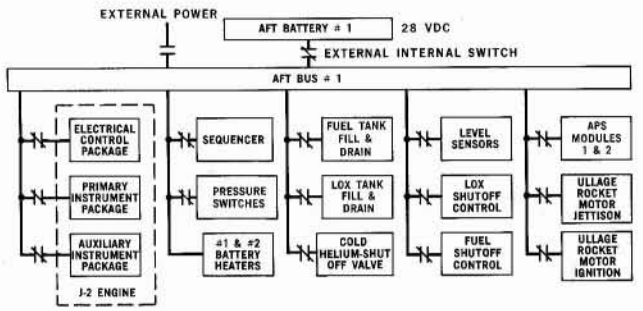
D-PB-121

Forward Power System No. 1



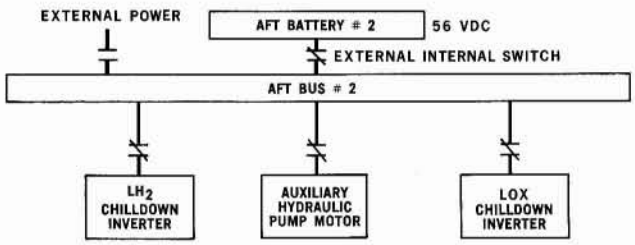
D-PB-122

Forward Power System No. 2



D-PB-123

Aft Power System No. 1



D-PB-124

Aft Power System No. 2

Forward Power System No. 2 includes a 28 vdc battery and power distribution equipment for the PU assembly, inverter-converter, and range safety system No. 2.

Aft Power System No. 1 includes a 28 vdc battery and power distribution equipment for the J-2 engine, pressurization systems, APS modules, TM signal power, aft battery heaters, hydraulic system valves, and stage battery sequencer.

Aft Power System No. 2 includes a 56 vdc battery and power distribution equipment for the auxiliary hydraulic pump, oxidizer chilldown inverter, and fuel chilldown inverter.

Silver-oxide, zinc batteries used for electrical power and distribution systems are manually activated. The batteries are "one-shot" units, and not interchangeable due to different load requirements.

Electrical power and distribution systems are activated by command through the aft umbilical prior to liftoff.

Telemetry and Instrumentation System

Radio telemetry is used for transmission of stage instrumentation information to ground receiving stations. Five transmitters, using two separate antenna systems, are capable of returning information on 60 continuous output data channels and 81 sampled data channels during S-IVB flight. Three different modulation systems are utilized: Pulse Amplitude Modulated/FM/FM (PAM/FM/FM); Single Sideband/FM (SS/FM); and Digital Data Acquisition System (DDAS).

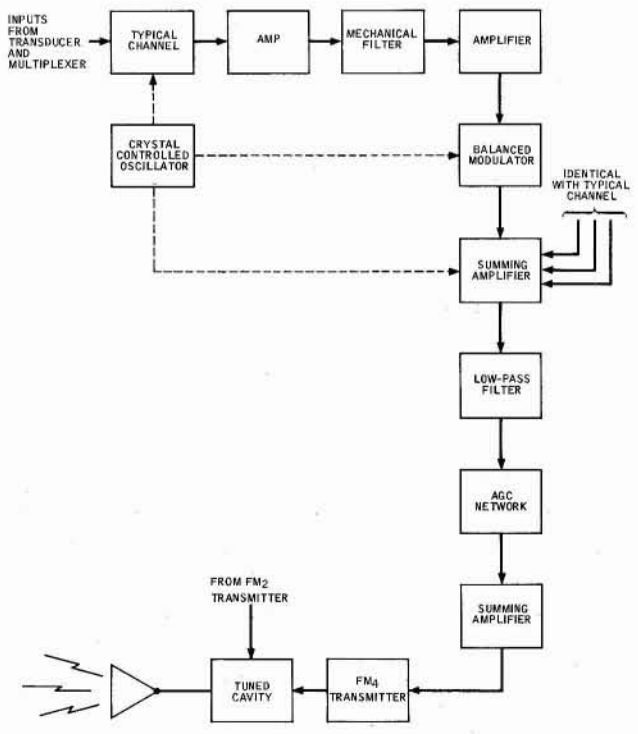
A DDAS airborne tape recorder stores sampled data normally lost during over-the-horizon periods of orbital missions, and plays back information when in range of ground stations.

PAM/FM/FM SYSTEMS

Transducer input signals constitute the PAM input. The PAM systems use an electronically switched mixing network that samples up to 30 channels of transducer inputs at 120 times a second. Deviations in transducer input voltages are represented as output pulses of varying amplitude for subsequent evaluation.

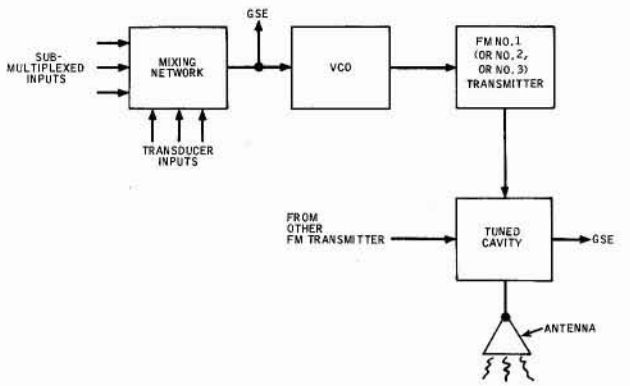
SS/FM SYSTEM

The SS/FM system is reserved for pertinent research requirements. Sonic vibration and acoustical data needed for manned flight development will be transmitted by this system.



D-PB-129

Basic SS/FM System



D-ORM-165

Basic PAM/FM/FM System

DIGITAL DATA ACQUISITION SYSTEM

The DDAS is used during automatic checkout on the ground, and for information playback from the airborne tape recorder when the stage is within communication range (line of sight) of ground stations. Redundant data is recorded in parallel from the PAM inputs and played back at high speed upon ground command. Continuous data flow charts are then pieced together at ground stations.

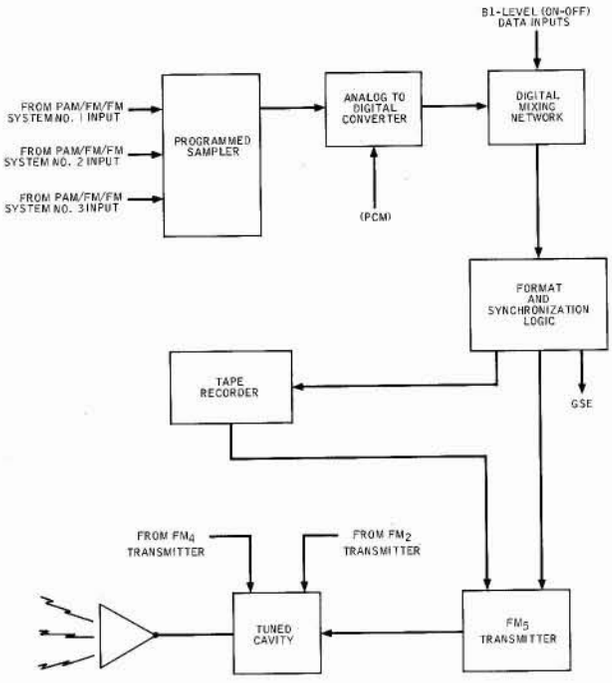
Environmental Control Systems

AFT SKIRT AND INTERSTAGE CONTROL THERMOCONDITIONING

The thermoconditioning and purge system purges the aft skirt and aft interstage of oxygen and combustible gases and distributes temperature controlled air or gaseous nitrogen around electrical equipment in the aft skirt.

The purging gas, supplied from a ground source through the umbilical, passes over the electrical equipment and flows into the aft interstage area. Some of the gas is directed through each of the auxiliary propulsion modules and exhausts into the interstage. A duct from the skirt manifold directs air or GN_2 to a thrust structure manifold. From the thrust structure manifold supply duct, a portion of air or GN_2 is directed to a shroud covering the hydraulic accumulator reservoir.

Temperature control is accomplished by two dual-element thermistor assemblies located in the gaseous exhaust stream of each of the auxiliary propulsion modules. Elements are wired in series to sense average temperature. Two series circuits are formed, each circuit utilizing one element from each thermister assembly. One series is used for temperature control, the other for temperature recording.



D-ORM-167

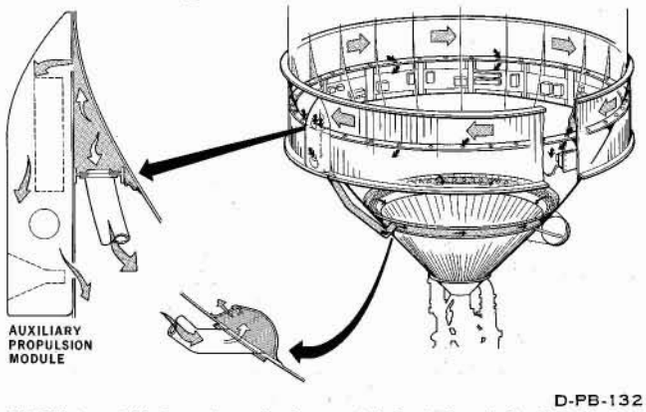
Basic PCM Digital Data Acquisition System

FORWARD SKIRT THERMOCONDITIONING

Electrical equipment in the S-IVB forward skirt area is thermally conditioned by a heat transfer system, using "cold plates" on which electronic components are mounted, and through which coolant fluid circulates. Coolant is pumped through the system from the IU and returned. Heat from electrical equipment attached to the cold plates is dissipated by conduction through the mounting feet and the cold plates to the fluid. Refer to the Instrument Unit section for a complete description of the IU environmental conditioning system.

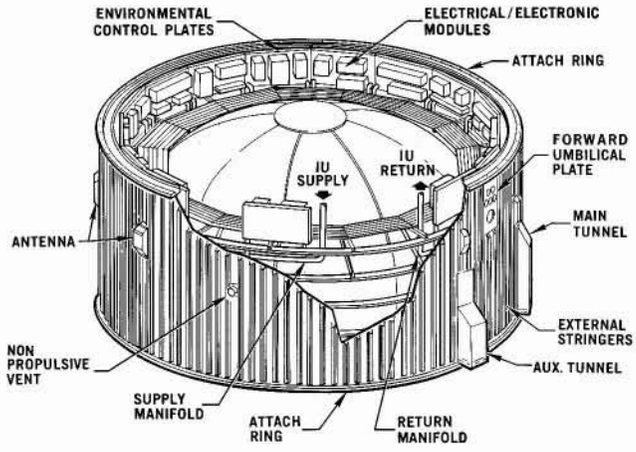
FORWARD SKIRT AREA PURGE

The forward skirt area is purged with gaseous nitrogren to minimize fire and explosion hazards while propellants are loaded or stored in the stage. Gaseous nitrogren is supplied and remotely controlled from a ground source.



D-PB-132

Aft Skirt and Aft Interstage Environmental Control — Indicating flow of nitrogen (or conditioned air) for cooling electrical equipment and purging combustionable gases from area.



D-PB-133

Forward Skirt Environmental Control System

Ordnance Systems

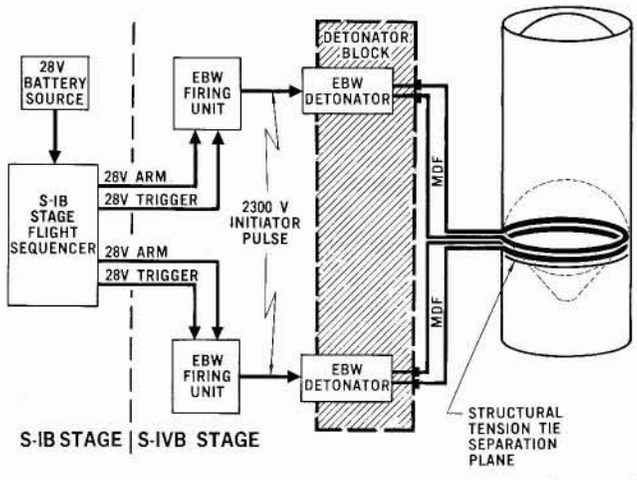
The Ordnance systems perform stage separation, retrorocket ignition, ullage control rocket ignition and jettison, and range safety functions.

STAGE SEPARATION SYSTEM

The stage separation system consists of a severable tension strap, mild detonating fuse (MDF), exploding bridgewire detonators (EBW), and EBW firing units.

The severable tension strap houses 2 redundant MDF cords in a "V" groove circumventing the stage between the aft skirt and aft interstage at the separation plane. Ignition of the MDF cords is triggered by a signal from the S-IB stage sequencer about 1 second after engine cutoff. The exploding bridgewire detonators (EBW), and EBW firing units.

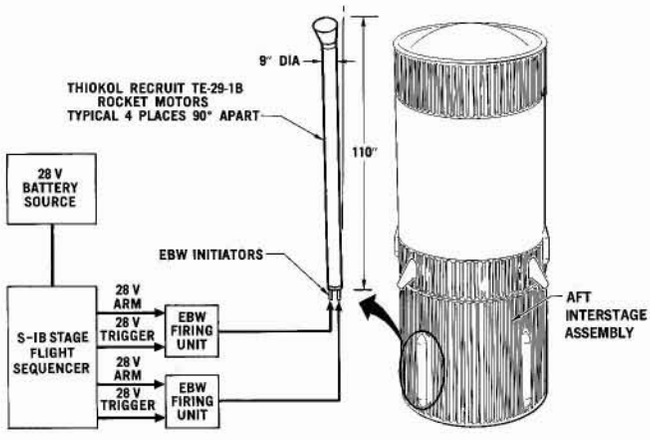
The MDF consists of a flexible metal sheath surrounding a continuous core of high explosive material. Once detonated, the explosive force of the MDF occurs at a rate of 23,000 feet per second.



D-PB-125

Separation System

The EBW detonator is electrically activated for initiating the MDF explosive train. A 2,300 vdc, 1,500 ampere pulse is applied to a small resistance wire and a spark gap. The high voltage electrical arc across the spark gap ignites a charge of high explosive material which in turn detonates the MDF. The high voltage and current requirement for ignition renders this system safe from random ground or vehicle electrical power. Upon command, each EBW firing unit supplies high voltage and current required to fire a specific EBW detonator.



Retrorocket System

RETROROCKET IGNITION SYSTEM

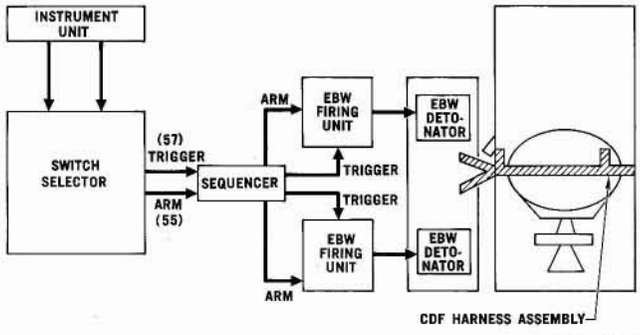
Four solid propellant retrorockets are mounted equidistant around the aft interstage assembly, and when ignited, assure clean separation of the S-IVB stage from the S-IB stage by decelerating or braking the spent booster. Each retrorocket is rated for a nominal thrust of 35,000 pounds, weight of 384 pounds, and burn time of about 1.5 seconds.

A signal from the S-IB stage initiates two EBW firing units located on the aft interstage. The EBW firing units ignite two detonator manifolds, which in turn ignite the retrorockets.

ULLAGE CONTROL ROCKET ENGINE AND JETTISON SYSTEM

Three solid propellant ullage rockets, located on the S-IVB aft skirt just forward of the stage separation plane, are ignited on signal from the stage sequencer by EBW initiators. The ullage rockets provide a positive "G" force to settle propellants in the tanks before ignition of the J-2 engine.

After firing, the burned-out ullage rocket casings and associated EBW firing units are jettisoned to reduce stage weight. Upon command from the stage sequencer, two forward and aft frangible nuts, which secure each rocket motor and its fairing to the stage, are detonated by confined detonating fuze (CDF), to free the entire assembly from the vehicle.



Ullage Rocket System

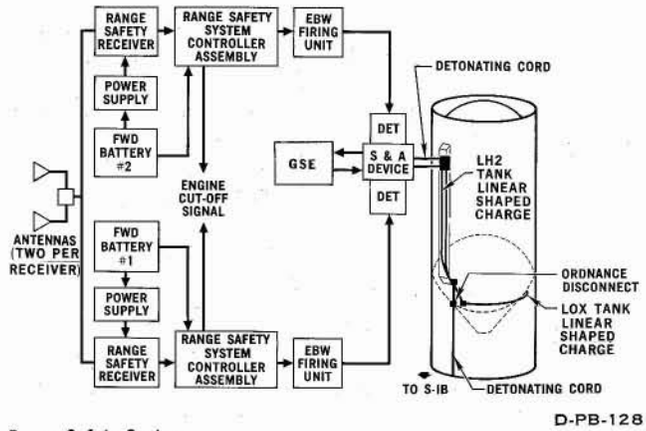
RANGE SAFETY SYSTEM

The range safety system terminates vehicle flight upon command of the range safety officer. Redundant systems are used throughout to provide maximum reliability.

Four antennae, mounted around the periphery of the S-IVB forward skirt assembly, feed two redundant range safety receivers located in the forward skirt assembly. Both receivers have separate power supplies and circuits. A unique combination of coded signals must be transmitted, received, and decoded to energize this destruct system.

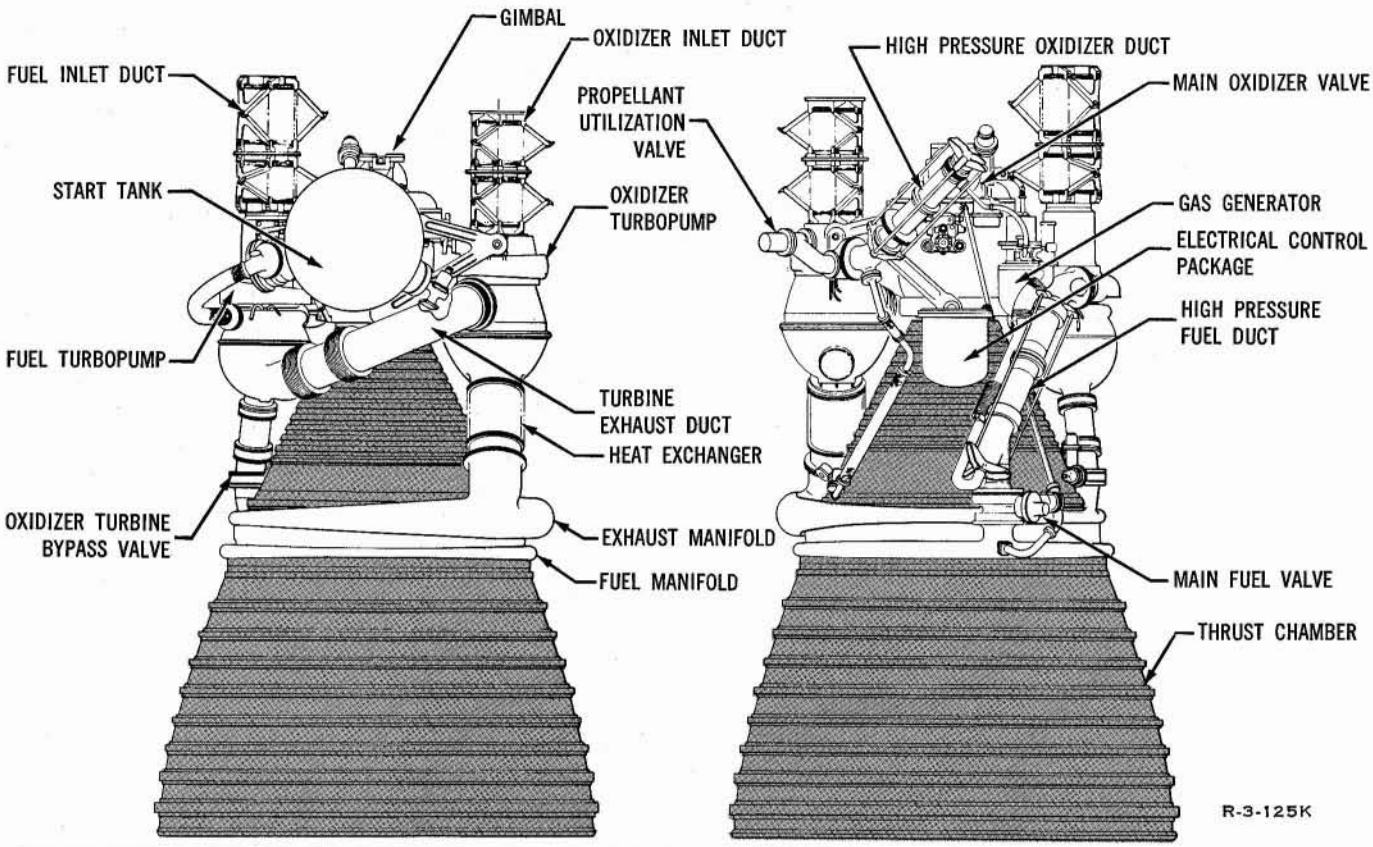
A safety and arming device prevents inadvertent initiation of the explosive train by providing a positive isolation of the EBW detonator and explosive train until arming is commanded. Visual and remote indications of SAFE and ARMED conditions are displayed at all times at the firing center. Upon proper command, EBW firing units activate EBW detonators.

A CDF, detonated by the safety and arming device, explodes a flexible linear-shaped charge which cuts through the tank skin to disperse both fuel and oxidizer.



Range Safety System

J-2 ENGINE FACT SHEET



LENGTH	11 ft 1 in
WIDTH	6 ft 8½ in
NOZZLE EXIT DIAMETER	6 ft 5 in ID
THRUST (altitude)	225,000 lb (maximum)
SPECIFIC IMPULSE (nominal)	426 sec
RATED RUN DURATION	500 sec
FLOWRATE: Oxidizer	451 lb/sec (2,865 gpm)
Fuel	82 lb/sec (8,340 gpm)
MIXTURE RATIO	5:1 oxidizer to fuel
NOMINAL CHAMBER PRESSURE (P_c)	709 psia
WEIGHT, DRY FLIGHT CONFIGURATION	3480 lb
EXPANSION AREA RATIO	27.5:1
COMBUSTION TEMPERATURE	5,750°F

Note: The J-2 engine is started at a 5:1 mixture ratio. After 5 seconds, the mixture ratio is changed to 5.5:1, 225,000 lb thrust, 423 seconds specific impulse, nominal, for about 325 seconds of flight. The mixture ratio is then changed to 4.7:1, 185,000 lb thrust, 427 seconds specific impulse, nominal, for the remainder of the flight.

J-2 ENGINE

J-2 ENGINE DESCRIPTION

The J-2 engine is a 225,000 pound thrust, high performance, upper stage, propulsion system, utilizing liquid hydrogen and liquid oxygen propellants, and incorporating a built-in capability for restart in flight.

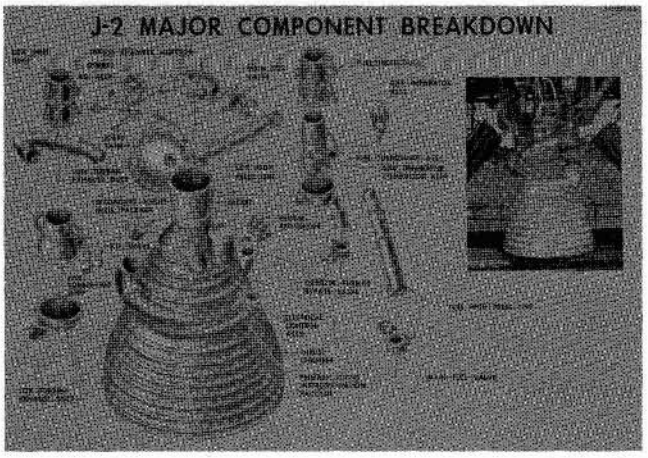
Major systems of the J-2 engine include a thrust chamber and gimbal system, propellant feed system, gas generator and exhaust system, electrical and pneumatic control system, start tank assembly system, and flight instrumentation system.

Thrust Chamber and Gimbal System

The J-2 engine thrust chamber serves as a mount for all engine components. It is composed of the following subassemblies: thrust chamber body, injector and dome assembly, gimbal bearing assembly, and augmented spark igniter. Thrust is transmitted through the gimbal mounted on the thrust chamber assembly dome to the vehicle thrust frame structure. The thrust chamber and injector receives the propellants from the turbopumps under pressure, mixes the propellants, and burns them to impart a high velocity to the expelled combustion gases to produce thrust for vehicle propulsion.

THRUST CHAMBER

The thrust chamber is constructed of stainless steel tubes of 0.012-inch wall thickness. Tubes with thin walls are required for heat transfer purposes. The thrust chamber tubes are stacked longitudinally and furnace brazed to form a single unit. The chamber is bell shaped with 27.5 to 1 expansion area ratio for efficient operation at altitude, and is regeneratively cooled by the fuel.



R-507-321-703

J-2 Major Component Breakdown

Fuel enters from a manifold to which it was delivered at a pressure of more than 1000 psi. It makes a one-half pass downward through 180 tubes and a full pass up through 360 tubes to cool the chamber.

DOME

The injector and dome assembly is located at the top of the thrust chamber. The dome manifolds the liquid oxygen and serves as a mount for the gimbal bearing and the augmented spark igniter.

INJECTOR

The thrust chamber injector is concentric orificed and porous faced. The purpose of the thrust chamber injector is to atomize and mix propellants in a manner to produce the most efficient combustion. Hollow oxidizer posts are machined to form an integral part of the injector. Fuel nozzles are threaded and installed over the oxidizer posts.

The injector face is formed from porous stainless steel and is welded at its periphery to the injector body. Each fuel nozzle is swaged to the face of the injector.

The injector receives liquid oxygen through the dome manifold and injects it through the oxidizer posts into the combustion area of the thrust chamber.

The fuel is received from the upper fuel manifold in the thrust chamber and injected through the fuel orifices which are concentric with the oxidizer orifices. The propellants are injected uniformly to ensure satisfactory combustion.

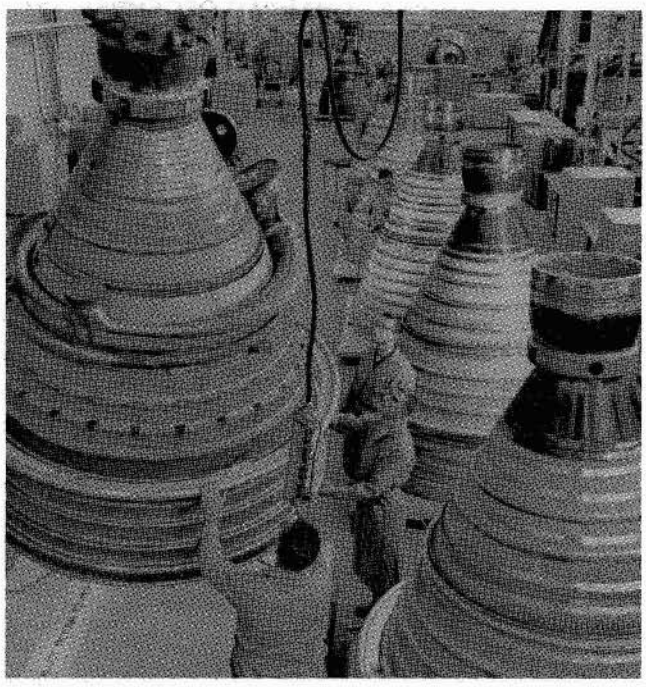
GIMBAL

The gimbal is a compact, highly-loaded (20,000 psi), universal joint consisting of a spherical, socket-type bearing with a Teflon-fiberglass composition coating that provides a dry, low-friction bearing surface. It also includes a lateral adjustment device for aligning the chamber with the vehicle.

The gimbal transmits the thrust from the injector assembly to the vehicle thrust structure and provides a pivot bearing for deflection of the thrust vector, thus providing flight attitude control of the vehicle. The gimbal is mounted on the top of the injector assembly.

AUGMENTED SPARK IGNITER

The augmented spark igniter (ASI) is mounted in the injector. It receives the initial flow of oxi-



R-MFG-622

Production Line—The production line for the hydrogen-fueled J-2 rocket engine at the Canoga Park, Calif. plant of Rocketdyne, a division of North American Aviation, Inc. The J-2 engine produces 225,000 lb. (maximum) of thrust for the S-IVB upper stage of the Saturn IB.

dizer and fuel, which is ignited by two spark plugs mounted in the side of the igniter chamber. When engine start is initiated, the spark exciters energize the spark plugs. Simultaneously, the control system starts the flow of oxidizer and fuel to the spark igniter. As the oxidizer and fuel enter the main combustion chamber, they are mixed and ignite.

Mounted in the ASI is an ignition monitor which indicates that proper ignition has taken place. The ASI operates continuously during entire engine firing, is uncooled, and is capable of multiple reignitions under all environmental conditions.

Propellant Feed System

The propellant feed system consists of separate oxidizer and fuel turbopumps, main fuel valve, main oxidizer valve, propellant utilization valve, oxidizer and fuel flowmeters, fuel and oxidizer bleed valves, and interconnecting lines.

OXIDIZER TURBOPUMP

The oxidizer turbopump is mounted on the thrust chamber diametrically opposite the fuel turbopump. It is a single-stage centrifugal pump with direct turbine drive. The oxidizer turbopump increases the pressure of the liquid oxygen and pumps it through high-pressure ducts to the thrust chamber at a rate of 2,900 gallons per minute. The pump operates at 8,600 rpm at a discharge pressure of 1,080 psia and develops about 2,200 brake horsepower. The pump and its two turbine wheels are mounted on a common shaft.

Power for operating the oxidizer turbopump is provided by a high-speed, two-stage turbine which is driven by the exhaust gases from a bi-liquid gas generator. The turbines of the oxidizer and fuel turbopumps are connected in series by exhaust ducting that directs the discharged exhaust gas from the fuel turbopump turbine to the inlet of the oxidizer turbopump turbine manifold. One static and two dynamic seals in series prevent the turbopump oxidizer fluid and turbine gas from mixing.

The turbopump operates in this manner: hot gas enters the nozzle and, in turn, the first-stage turbine wheel. After passing through the first-stage turbine wheel, the gas is redirected by the stator blades and enters the second-stage turbine wheel. The gas then leaves the turbine through exhaust ducting, passes through the heat exchanger, and exits through the thrust chamber. Power from the turbine is transmitted to the inducer and impeller by the pump shaft. The velocity of the liquid oxygen is increased through the inducer and impeller. As the liquid oxygen enters the outlet volute, velocity is converted to pressure and the liquid oxygen is discharged into the outlet duct at high pressure.

FUEL TURBOPUMP

The fuel turbopump, also mounted on the thrust chamber, is a turbine-driven, axial flow pumping unit consisting of an inducer, a seven-stage rotor, and a stator assembly. It is a high-speed pump operating at 27,000 rpm, and is designed to in-

R-MFG-635

J-2 Engine Assembly—Hydrogen-fueled J-2 rocket engines for upper stages of Saturn IB vehicles are completed on the Rocketdyne assembly line.

crease hydrogen pressure to 1,225 psia through high-pressure ducting to the thrust chamber. It develops 7,800 brake horsepower.

Power for operating the turbopump is provided by the high-speed, two-stage turbine. Hot gas from the gas generator is routed to the turbine inlet manifold which distributes the gas to the nozzle where it is expanded and directed at a high velocity into the first-stage turbine wheel.

After passing through the first-stage turbine wheel, the gas is redirected through the stator blades and enters the second-stage turbine wheel. The gas leaves the turbine through the exhaust ducting. Three Dynamic seals in series prevent the pump fluid and turbine gas from mixing. Power from the turbine is transmitted to the pump by means of a one-piece shaft.

Bearings in the liquid hydrogen and liquid oxygen turbopumps are lubricated by the fluid being pumped, as the extremely low operating temperature of the engine precludes use of lubricants or other fluids.

MAIN OXIDIZER VALVE

The main oxidizer valve is a butterfly-type valve, spring-loaded to the closed position, pneumatically operated to the open position, and pneumatically assisted to the closed position. It is mounted between the oxidizer high-pressure duct from the oxidizer turbopump and the oxidizer inlet on the thrust chamber assembly.

Pneumatic pressure from the normally closed port of the mainstage control solenoid valve is routed to both the first and second stage opening actuators of the main oxidizer valve. Application of opening pressure in this manner, together with controlled venting of the main oxidizer valve closing pressure provides a controlled ramp opening of the main oxidizer valve through all temperature ranges.

A sequence valve, located within the MOV assembly, supplies pneumatic pressure to the opening control part of the gas generator control valve and through an orifice to the closing part of the oxidizer turbine bypass valve.

MAIN FUEL VALVE

The main fuel valve is a butterfly-type valve, spring-loaded to the closed position, pneumatically operated to the open position, and pneumatically assisted to the closed position. It is mounted between the fuel high-pressure duct from the fuel turbopump and the fuel inlet manifold of the thrust chamber assembly. The main fuel valve controls the flow of fuel to the thrust chamber. Pressure from the ignition stage control valve on the pneumatic control package opens the valve during engine start. As the gate starts to open, it allows fuel to flow to the fuel inlet manifold.

PROPELLANT UTILIZATION VALVE

The propellant utilization valve is an electrically operated, two-phase, motor-driven, oxidizer-transfer valve and is located at the oxidizer turbopump outlet volute. The propellant utilization valve ensures the simultaneous exhaustion of the contents of the propellant tanks. During engine operation, propellant level sensing devices in the vehicle propellant tanks control the valve gate position for adjusting the oxidizer flow to ensure simultaneous exhaustion of fuel and oxidizer.

The propellant utilization (PU) valve and its servomotor are supplied with the engine. A position feedback potentiometer is also supplied as a part of the PU valve assembly. The PU valve assembly and a S-IVB stage or a facility-mounted control system make up the propellant utilization system.

OXIDIZER AND FUEL FLOWMETERS

The oxidizer and fuel flowmeters are identical helical-vaned, rotor-type flowmeters. They are located in the oxidizer and fuel high-pressure ducts. The flowmeters measure propellant flowrates in the high-pressure propellant ducts. The four-vane rotor in the hydrogen system produces four electrical impulses per revolution and turns approximately 3,700 revolutions per minute at nominal flow. The six-vane rotor in the liquid oxygen system produces six electrical impulses per revolution and turns at approximately 2,600 revolutions per minute at nominal low.

PROPELLANT BLEED VALVE

The propellant bleed valves used in both the oxidizer and fuel systems are poppet-type which are spring loaded to the open position and pressure actuated to the closed position. Both propellant bleed valves are mounted to the bootstrap lines adjacent to their respective turbopump discharge flanges.

The valves allow propellant to circulate in the lines to achieve proper operating temperature prior to engine start. The bleed valves are engine controlled. At engine start, a helium control solenoid valve in the pneumatic control package is energized allowing pneumatic pressure to close the

bleed valves, which remain closed during engine operation.

Gas Generator and Exhaust System

This system consists of the gas generator, gas generator control valve, turbine exhaust system and exhaust manifold, heat exchanger, and oxidizer turbine bypass valve.

GAS GENERATOR

The gas generator produces hot gases to drive the oxidizer and fuel turbines and consists of a combustor containing two spark plugs, a control valve containing oxidizer and fuel ports, and an injector assembly.

When engine start is initiated, the spark exciters in the electrical control package are energized providing energy to the spark plugs in the gas generator combustor. Propellants flow through the control valve to the injector assembly and into the combustor where they are mixed and burned. The resulting hot gases pass through the combustor outlet and are directed to the fuel turbine and then to the oxidizer turbine.

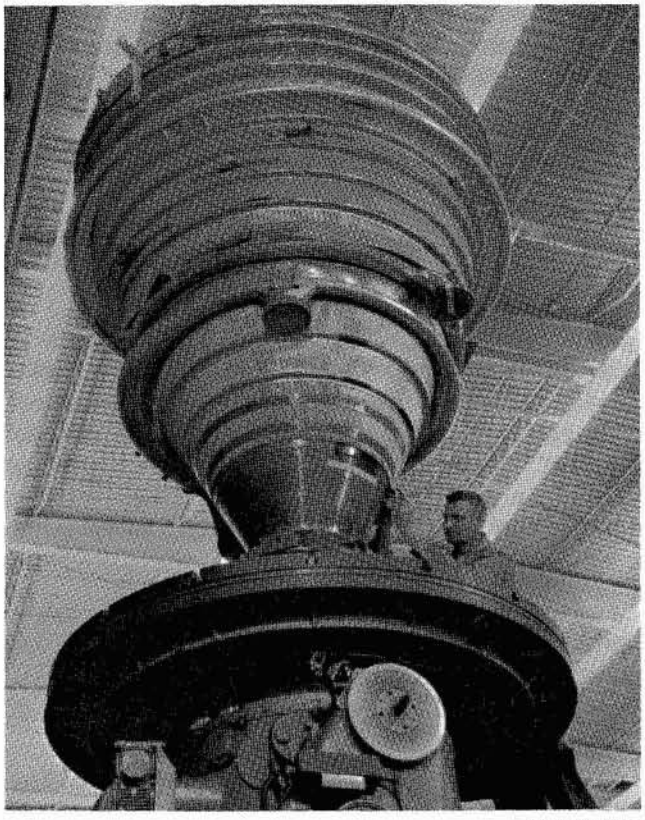
GAS GENERATOR CONTROL VALVE

The gas generator control valve is a pneumatically operated poppet type that is spring-loaded to the closed position. The oxidizer and fuel poppets are mechanically linked by an actuator. The gas generator control valve controls the flow of propellants through the gas generator injector.

When the mainstage signal is received, pneumatic pressure is applied against the gas generator control valve actuator assembly which moves the piston and opens the fuel poppet. During the fuel poppet opening, an actuator contacts the piston that opens the oxidizer poppet. As the opening pneumatic pressure decays, spring loads close the poppets.

TURBINE EXHAUST SYSTEM

The turbine exhaust ducting and turbine exhaust hoods are of welded sheet metal construction. Flanges utilizing dual (Naflex) seals are used at component connections. The purpose of the exhaust ducting is to conduct turbine exhaust gases to the thrust chamber exhaust manifold which encircles the thrust chamber approximately halfway between the throat and the nozzle exit. Exhaust gases pass through the heat exchanger and exhaust into the main thrust chamber through 180 triangular openings between the tubes of the thrust chamber.



R-MFG-620

Optical Alignment — Points where component parts will be joined to thrust chamber of J-2 engine are precisely located by optical tools on production line.

HEAT EXCHANGER

The heat exchanger is a shell assembly, consisting of a duct, bellows, flanges, and coils. It is mounted in the turbine exhaust duct between the oxidizer turbopump and the thrust chamber. It heats and expands helium gas or converts liquid oxygen to gaseous oxygen for maintaining vehicle oxidizer tank pressurization. During engine operation, either liquid oxygen is tapped off the oxidizer high-pressure duct or helium is provided from vehicle stage and routed to the heat exchanger coils. As turbine exhaust gases pass over the heat exchanger coils, the liquid oxygen is converted to gaseous oxygen. The gaseous oxygen or helium is then routed to the vehicle oxidizer tank to maintain vehicle propellant tank pressurization.

OXIDIZER TURBINE BYPASS VALVE

The oxidizer turbine bypass valve is a normally open, spring-loaded, gate type. It is mounted in the oxidizer turbine bypass duct. The valve gate is equipped with a nozzle whose size is determined during engine calibration. The valve prevents an over-speed condition of the oxidizer turbopump and acts as a calibration device for the turbopump performance balance.

Control System

The control system includes a pneumatic system and a solid-state electrical sequence controller packaged with spark exciters for the gas generator and the thrust chamber spark plugs, plus interconnecting electrical cabling and pneumatic lines.

PNEUMATIC SYSTEM

The pneumatic system consists of a high pressure helium controlled gas storage tank, a regulator to reduce the pressure to a usable level, and electrical solenoid control valves to direct the central gas to the various pneumatically controlled valves.

ELECTRICAL SEQUENCE CONTROLLER

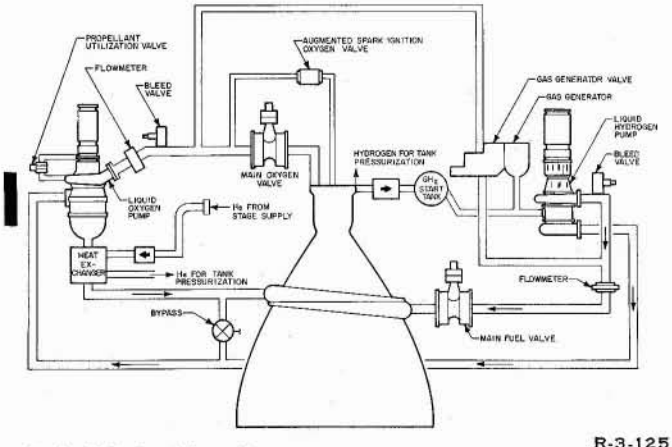
The electrical sequence controller is a completely self-contained, solid-state system, requiring only dc power and start and stop command signals. Pre-start status of all critical engine control functions is monitored in order to provide an "engine ready" signal. Upon obtaining "engine ready" and "start" signals, solenoid control valves are energized in a precisely timed sequence to bring the engine through ignition, transition, and into mainstage operation. After shutdown, the system automatically resets for a subsequent restart.

Start Tank Assembly System

This system is made up of an integral helium and hydrogen start tank, which contains the hydrogen and helium gases and the valves for starting and operating the engine.

HELIUM AND HYDROGEN TANKS

The spherical helium tank is positioned inside the hydrogen tank to minimize engine complexity. It holds 1000 cubic inches of helium. The larger spherical hydrogen gas tank has a capacity of 7257.6 cubic inches. Both tanks are filled from a ground source prior to launch.



R-3-125

Basic J-2 Engine Schematic

Flight Instrumentation System

The J-2 engine is provided with an instrumentation system which measures basic pressure, temperature, flow, speed and position parameters with the capability of transmitting these signals to a ground recording system or a telemetry system, or both. The instrumentation system is designed for use throughout the life of the J-2 engine, including the first static acceptance firing tests. The total instrumentation system consists of two independent primary and auxiliary packages. The primary package measures those parameters considered critical to all engine static firings and vehicle flights.

AUXILIARY PACKAGE

An auxiliary package is designed for use during early vehicle flights. It can be removed from the engine without interfering with the operation of the primary system. It contains sufficient flexibility to provide for deletion, substitution, or addition of parameters so that design changes requiring modifications will be minimized.

ENGINE OPERATION

Start Sequence

Start sequence is initiated by supplying energy to two spark plugs in the gas generator and to the augmented spark igniter to the thrust chamber for ignition of the propellants. Next, two solenoid valves are actuated: one for helium control, and one for ignition phase control. Helium is routed to hold the propellant bleed valves closed and to purge the thrust chamber LOX dome, the LOX pump intermediate seal, and the gas generator oxidizer passage. In addition, the main fuel and augmented spark igniter oxidizer valves are opened. An ignition flame is thus created in the center of the thrust chamber injector. A start tank discharge valve is then opened to initiate turbine spin. After a short interval, this valve is closed and a mainstage control solenoid is actuated to: (1) turn off gas generator and thrust chamber helium purges ;(2) open the gas generator control valve (hot gases from the gas generator now drive the pump turbines); (3) open the main oxidizer valve to the first position (14 degrees); (4) close the oxidizer turbine bypass valve (a portion of the gases for driving the oxidizer turbopump were bypassed during the ignition phase); (5) gradually bleed the pressure from the closing side of the oxidizer valve pneumatic actuator controlling the slow opening of this valve for smooth transi-

tion into mainstage. Energy in the spark plugs is cut off and the engine is operating at rated thrust. Within several seconds, the gaseous hydrogen tank will be recharged in those engines having a restart requirement. High pressure hydrogen gas is tapped from the fuel manifold located at the top of the thrust chamber to repressurize the tank.

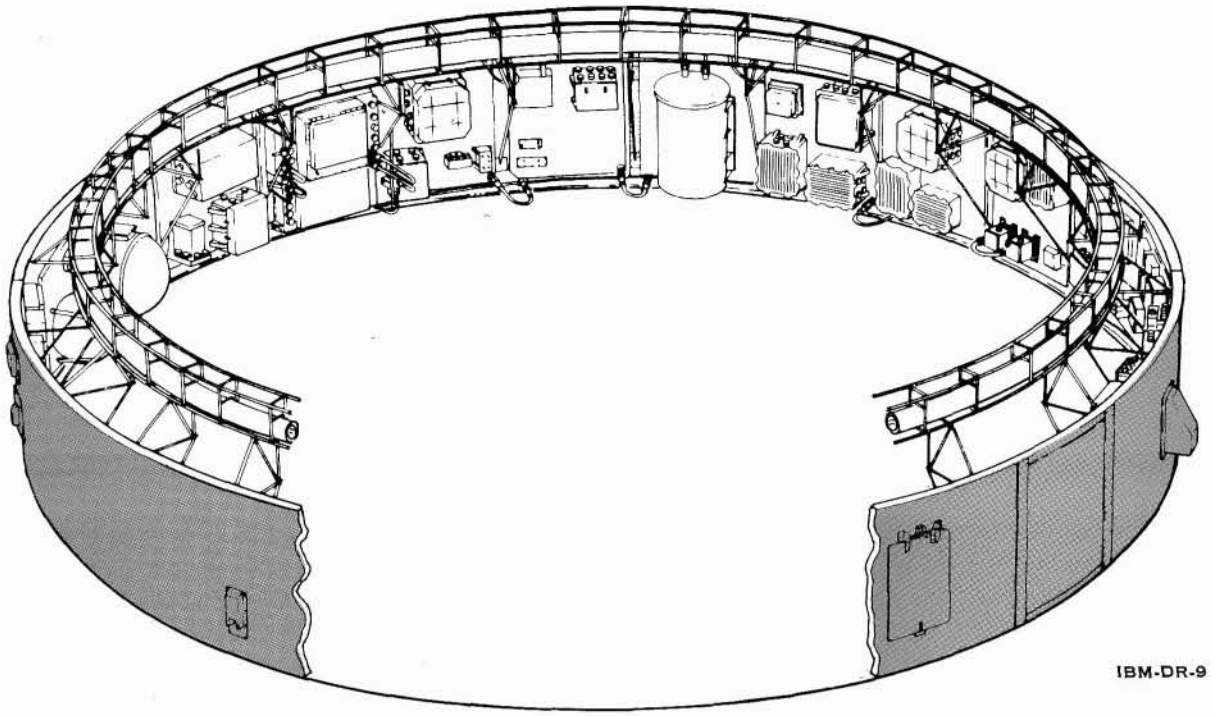
Flight Mainstage Operation

During mainstage operation, engine thrust may be varied between 175,000 and 225,000 pounds by actuating the propellant utilization valve to increase or decrease oxidizer flow. This is beneficial to some flight trajectories where overall mission performance is desired to make greater payloads possible.

Cutoff Sequence

When the engine cutoff signal is received by the electrical control package, it de-energizes the mainstage and ignition phase solenoid valves and after a short delay, de-energizes the helium control solenoid. This, in turn, closes the fuel valve, oxidizer valve, gas generator control valve, and augmented spark igniter valve. The oxidizer turbine bypass valve and propellant bleed valves open to complete the engine cutoff sequence.

INSTRUMENT UNIT FACT SHEET



IBM-DR-9

DIAMETER: 260 in
HEIGHT: 36 in
WEIGHT: 4,100 lb (approx)

MAJOR SYSTEMS

ENVIRONMENTAL CONTROL SYSTEM: Provides cooling for electronic modules and components within the IU and forward compartments of S-IVB stage.

GUIDANCE AND CONTROL SYSTEM: Determines course of Saturn IB through space, and adapts that course to fulfill mission requirements.

MEASURING AND TELEMETRY SYSTEM: Measures vehicle conditions and reactions during mission, and transmits this information to ground for subsequent analysis.

RADIO FREQUENCY AND TRACKING SYSTEM: Provides for ground station-to-vehicle communication.

ELECTRICAL SYSTEM: Provides basic operating power for all electronic and electrical equipment in the IU.

EMERGENCY DETECTION SYSTEM: Monitors vehicle performance, and may initiate automatic mission abort if an emergency arises.

INSTRUMENT UNIT

INSTRUMENT UNIT DESCRIPTION

The Instrument Unit (IU) for Saturn IB was designed by NASA at MSFC, and was developed from the Saturn I IU. Overall responsibility for the IU is being assigned to IBM Federal Systems Division for fabrication and assembly of the unit, system testing, and integration and checkout of the unit with the launch vehicle.

IBM also assembles and delivers "software" necessary to support the IU. Software packages include computer programs necessary to simulate mission conditions and predict vehicle performance, operate automated checkout equipment, guide the vehicle in flight, and post-flight reduction and analysis of vehicle environment and performance data.

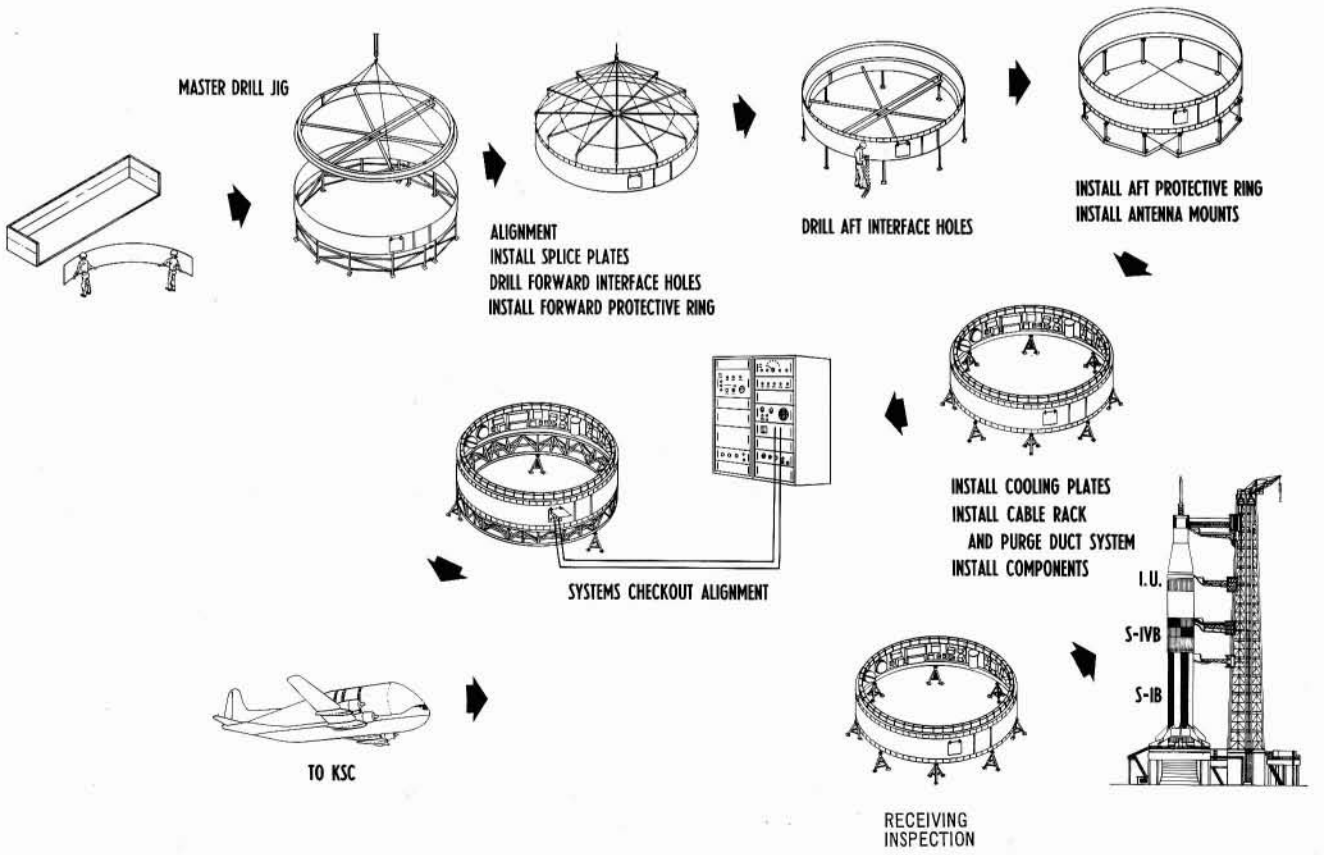
The IU is the nerve center of the Saturn IB. It contains electronic and electrical equipment necessary for guidance, tracking, and origination and communication of vehicle environmental and performance data. The IU also contains environmental control equipment for heat dissipation, and batteries and power supplies which furnish basic operating power for electronic equipment.

The stage structure is 260 inches in diameter and 36 inches high, and becomes a load-bearing part of the vehicle. It supports the components within the IU, and the weight of the spacecraft.

INSTRUMENTATION UNIT FABRICATION AND ASSEMBLY

The structure is manufactured in three 120 degree segments, each consisting of thin-wall aluminum alloy face sheets bonded over a core of aluminum honeycomb. An aluminum alloy channel ring, bonded to the top and bottom edge of each segment, provides mating surfaces between the IU, the S-IVB stage, and the payload adapter. Brackets mounted on the inner skin of each segment provide attachment points for the cold plates of the environmental control system, or for components not suitable for cold plate installation.

Segments are precisely aligned and joined by means of splice plates bolted to the inner and outer surface of each joint. This spliced-joint design provides a structure of great strength that may be easily disassembled for packaging and shipping. A spring-loaded umbilical door in the structure provides for electrical connections



IU Production Sequence

IBM DR 22

Structure Segments — Prior to splicing, mounting brackets for thermal conditioning panels can be seen on interior surface of segments. The exterior of the spring-loaded umbilical door and the access door are visible at right center.

between equipment within the IU and ground test areas. A larger access door, bolted in place, permits personnel to enter the IU after it is mated to the launch vehicle.

Manufacture of an IU begins with the arrival of three curved structural segments three feet high by 14 feet long at IBM's Huntsville, Alabama, facility. Each segment weighs only 140 pounds.

Extremely accurate theodolites similar to a surveyor's transit are used to precisely align the segments in a circle prior to splicing. Metal splicing plates are attached at each of the three joints, and the holes which permit the IU to be joined to mating surfaces of the launch vehicle are drilled at top and bottom edges of the structure. Protective rings are bolted to these edges to stiffen the structure. Following completion of splices, holes are cut through the structure for vehicle antennas.

When fabrication of the structure is completed, IU module and component assembly operations are started. Temperature transducers are fastened to the inner skin; thermal conditioning panels (cold plates) are then mounted, and a cable tray is installed around the top of the structure. Components are mounted on the thermal conditioning panels, and thermal conditioning system pumps, storage tanks (called accumulators), heat exchangers, and plumbing are installed. A nitrogen supply system is installed for gas bearings of the inertial platform and pressurization of the environmental control system. Finally, ducts, tubing, and electrical cables complete the assembly and the IU now weighing in excess of 4,000 pounds is ready for a long series of tests.

INSTRUMENTATION UNIT SYSTEMS

Environmental Control System

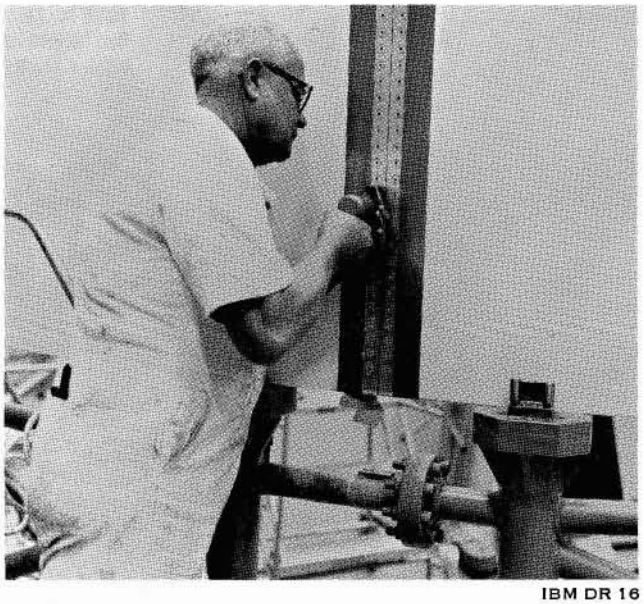
The environmental control system provides for cooling electronic modules and components within the IU and the forward compartment of the S-IVB stage through 32 thermal conditioning panels, or cold plates, secured to the inner skin of the IU and S-IVB skirt.

Coolant from a reservoir within the IU is circulated through the cold plates to provide heat transfer from electronic components mounted on the plates. The coolant is a solution of 60 per cent methanol and 40 per cent water.

Prior to liftoff, heat is removed from the coolant by a preflight heat exchanger serviced by ground support equipment. After launch vehicle attains altitude, heat dissipation is accomplished by a sublimator-heat exchanger.

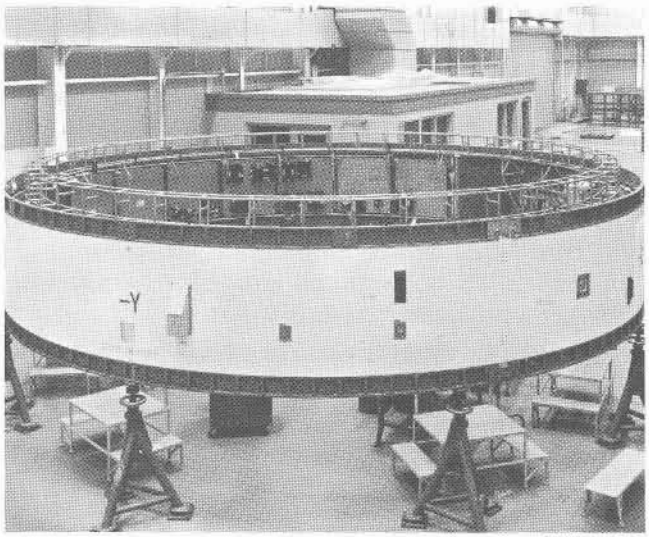
In the sublimation process, water is passed through porous plates and the sublimator side of the heat exchanger, which are exposed to the low temperature and pressure of outer space and freezes, blocking the pores of the plate. Heat from the coolant is transferred through the plate to be absorbed by the ice, which sublimes to water vapor.

The system is self-regulating in that the rate of heat dissipation varies directly with the amount of heat input, speeding up or slowing down as more or less heat is generated. If the coolant temperature falls below a pre-set level, an electronically controlled valve causes the coolant mixture to bypass the sublimator until the temperature rises sufficiently to require cooling again.



IBM DR 16

Splice Joint Operations — Final grinding of a splice joint ensures a smooth surface prior to splice plate assembly.

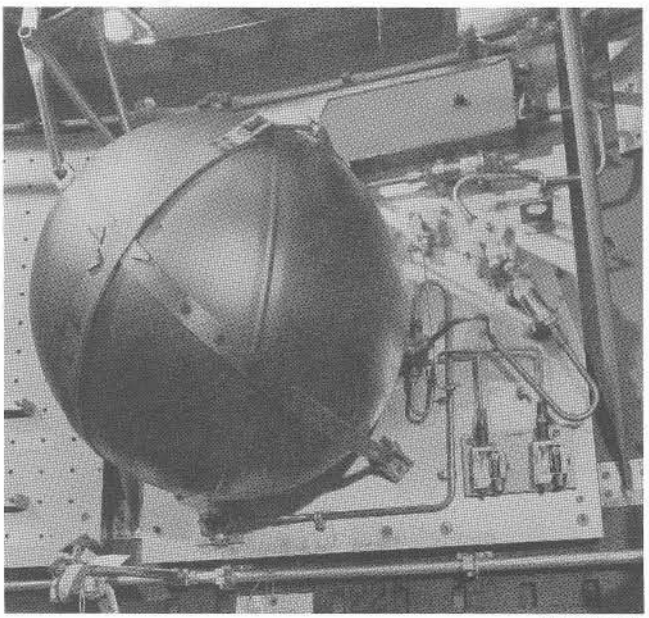


IBM DR 19

Instrument Unit Assembly in IBM Manufacturing Area — Splicing operations and assembly of the tubular cable tray are complete, the cold plates have been installed, and installation of components is underway. Forward and aft protective rings will remain in place until the unit is mounted on the launch vehicle. Objects protruding from the outer skin are flight antennas.

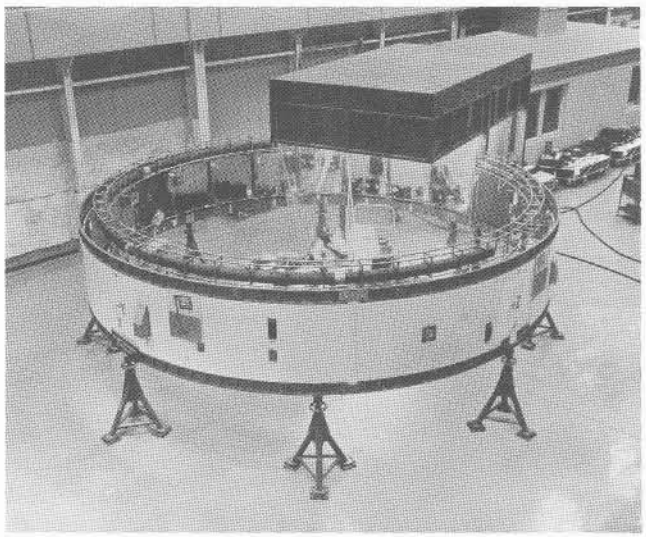
A supply of nitrogen gas is provided to maintain artificial pressure in the reservoirs for both the coolant solution and sublimator water during periods of weightless flight. Nitrogen from this supply is also provided to the gas bearing of the ST-124-M inertial platform.

A pump circulates the coolant, and the necessary valves and piping to control its flow complete the environmental control equipment.



IBM DR 21

GN_2 Storage Sphere — In place next to the ST-124-M inertial platform, the sphere holds 2 cubic feet of gas used for gas bearings of the platform. Also visible are a pressure regulator, heat exchanger for warming gas, and pressure indicators.



IBM DR 23

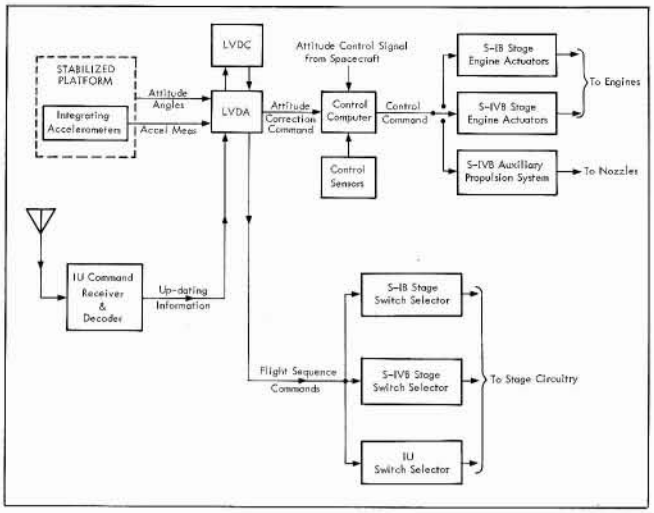
Instrument Unit — A mobile clean room protects against contamination during assembly of environmental control system components. Gaseous nitrogen will be circulated from a ground supply through the duct partially assembled in the cable tray to purge the IU following vehicle fueling.

Guidance and Control System

The guidance and control system navigates by determining vehicle position and velocity from measurements made on board, guides by computing maneuvers necessary to achieve the desired end conditions of trajectory and controls by executing maneuvers to control the proper hardware. These functions are performed in a manner to accomplish mission success with a minimum of propellant consumption. In addition, the system avoids excessive structural loads caused by aerodynamic flow and wind during ascent through the dense portion of the atmosphere, and during vacuum flight controls maneuvers to assure that structural bending and propellant sloshing are not dangerously excited by maneuvers.

Three major components or subsystems comprise the system: an inertial platform (ST-124-M); a high speed digital computer, and an analog control computer.

Prior to launch, parameters that define a particular launch window are fed into the launch vehicle digital computer (LVDC) and the ST-124-M platform is erected to a true vertical and aligned to launch azimuth. Just prior to liftoff the platform is released and becomes space fixed oriented. Vehicle acceleration is sensed and measured to be integrated with time since launch. These integrated inputs are used to determine vehicle position to its starting point and intended destination, burn time for engines, and direction of thrust required to arrive at the proper altitude and velocity at the required point in space.



IBM DR 8

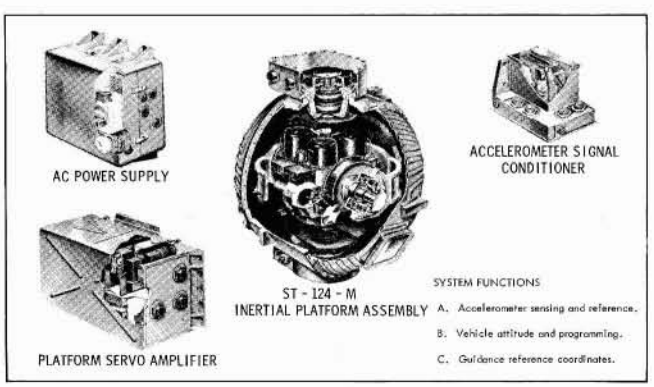
Block Diagram of Guidance and Control System

GUIDANCE AND CONTROL FUNCTIONS

The flight control computer generates the proper control commands for the engine actuators and the S-IVB auxiliary propulsion system. This is accomplished by processing and combining signals from the LVDC, control sensors, and spacecraft.

During first stage powered flight, the vehicle lifts off vertically from the launch pad and maintains its liftoff orientation long enough to clear the ground equipment. It then performs a roll maneuver to align the vehicle with flight azimuth direction. This maneuver gives the vehicle control axes the correct alignment to the flight plane thus simplifying the computations in the attitude control loop. On the launch pad, the vehicle always has a roll orientation fixed to the launching site.

During first-stage propulsion, a time tilt program, stored in the LVDC, is applied simultaneously with the described roll maneuver. The pitch angle of the vehicle is commanded according to the tilt program which is a function of time only and is independent of navigation measurements. However, navigation measurements and computations are performed throughout the flight, beginning at the time the platform is released (i.e., 5 seconds before liftoff). Cutoff of the first stage engines occurs when the fuel level in the tanks reaches a predetermined level. Thereafter, the first stage is separated from the launch vehicle.

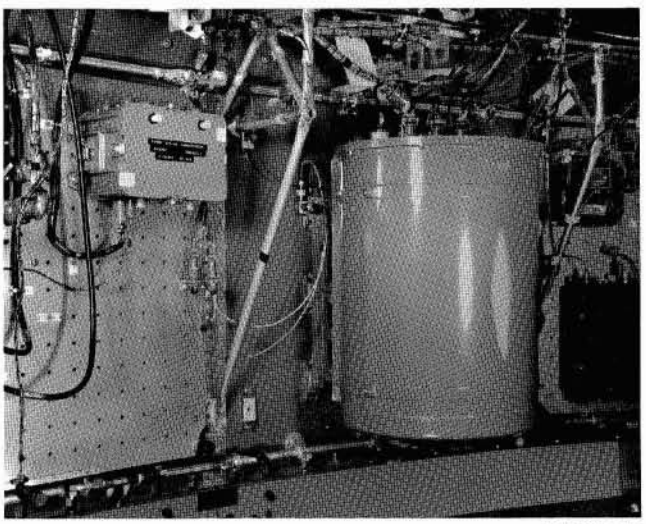


IBM DR 4

ST-124-M Inertial Platform System

After ignition of the S-IVB stage, adaptive guidance (i.e., the iterative guidance mode) is used during all propellant flight phases of the mission. The iterative guidance mode computes the pitch and yaw angle of the required direction to guide the vehicle on a minimum propellant trajectory into the predetermined orbit.

During orbital coast flight, the navigation program continually computes the vehicle position and velocity from the equations of motion based on insertion conditions. Attitude of the vehicle roll axis in orbit is maintained at 90 degrees with respect to the local vertical. The local vertical is determined from navigational computation.



IBM DR 17

Instrument Unit Interior During Assembly — The large, cylindrical component simulates size and shape of the flight control computer, and is used to check cable lengths and mounting arrangement. Use of nonflight hardware such as this during initial assembly minimizes handling of critical flight components.

On orbit, navigation and guidance information in the LVDC can be updated by data transmission from ground stations through the IU radio command system.

TRIPLE RELIABILITY

To ensure the accuracy and reliability of information supplied by the guidance and control system, critical circuits in the LVDC are provided in triplicate. Known as triple modular redundancy (TMR), this system corrects for failure or inaccuracy of a particular circuit by providing three identical circuits. Each circuit produces an output which is then voted upon. In case of a discrepancy in these outputs, the majority rules, and a random failure or error can be ignored. In addition, the computer memory

is duplexed and if an error is found in one portion of the memory, the required output is obtained from the other memory and correct information read back into both memories, thus correcting the error.

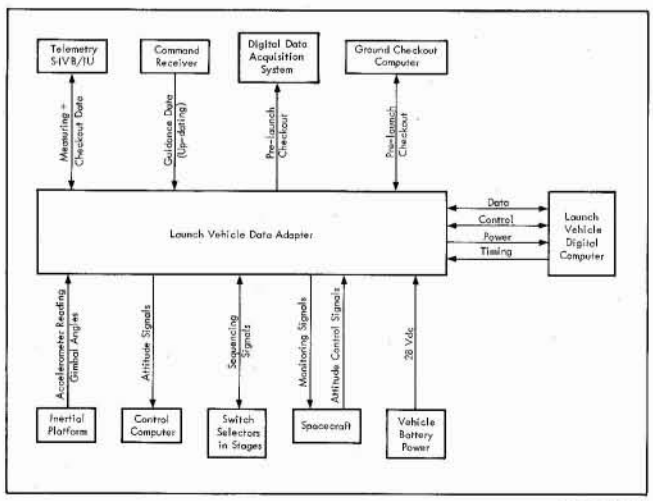
The ST-124-M inertial platform provides signals representing vehicle attitude. Since a small error in these signals could produce an intolerable error in ultimate vehicle position, friction in these components must be held to an absolute minimum. To this end, their bearings are floated in a thin film of dry nitrogen supplied at a controlled pressure and flow rate from reservoirs within the IU.

PRELAUNCH FUNCTIONS

In addition to guidance computations, other functions are performed by the launch vehicle digital computer and the launch vehicle data adapter (LVDA). During prelaunch, the units conduct test programs; during launch phase they direct engine ignition and cutoff, stage separations, and conduct reasonableness tests of vehicle performance. During earth orbit, the computers direct attitude control, conduct tests, isolate malfunctions, and control transmission of data.

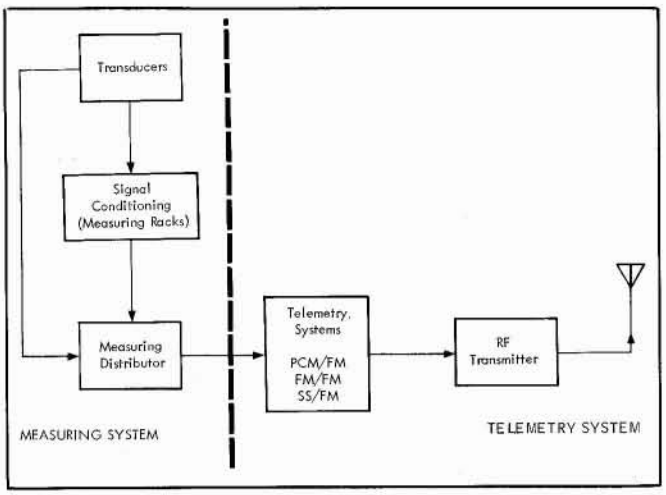
Measuring and Telemetry System

A basic requirement for vehicle performance evaluation and for planning of future missions is a knowledge of conditions encountered during each mission, and vehicle reaction to those conditions. This requirement is satisfied by the IU's measuring and telemetry system. Measuring sensors, or transducers, are located throughout the vehicle to monitor environment and systems' performance.



IBM DR 6

Block Diagram of Guidance and Control System — The guidance and control system's LVDA and LVDC receive information from all parts of the vehicle and issue commands based on the evaluation of incoming data in accordance with previously stored instructions.



IBM DR 1

Measuring and Telemetry System

Measurements are made of physical quantities such as mechanical movements, atmospheric pressures, sound levels, temperatures, and vibrations, and are transformed into electrical signals. Measurements are also made of existing electrical signals, such as voltage, currents, and frequencies which are used to determine sequence of stage separation, engine cutoff, and other flight events, and to determine performance of onboard equipment.

Approximately 200 measurements are made within the IU. A wide variety of sensors are employed to obtain different types of information required: acoustic transducers monitor sound levels; resistor or thermistor transducers monitor temperature environments; bourdon-tube or bellows transducers measure pressures; force-balance, or piezoelectric accelerometers measure force levels at critical points; flow meters determine rates of fluid flow.

Various measuring devices produce a variety of outputs, and before these outputs can be effectively utilized, they must be standardized to some extent. Signal conditioning modules are employed to adapt transducer outputs within a uniform range of 0-5 volts dc.

Different types of data require different modes of transmission, and the telemetry portion of the system provides three such modes: SS/FM, FM/FM, and PCM/FM. Each type of information is routed to the most efficient telemetry equipment; a routing performed by the measuring racks within the IU.

To provide for maximum utilization of transmission equipment, multiplexing is employed on all telemetry channels. Information originated by various measuring devices is repeatedly sampled by multiplexers, or commutators, and successive

samples from different sources are transmitted to earth.

Information transmitted over any channel represents a series of measurements made at different points within the vehicle, and this time sharing permits a large amount of data to be handled with a minimum of communication equipment.

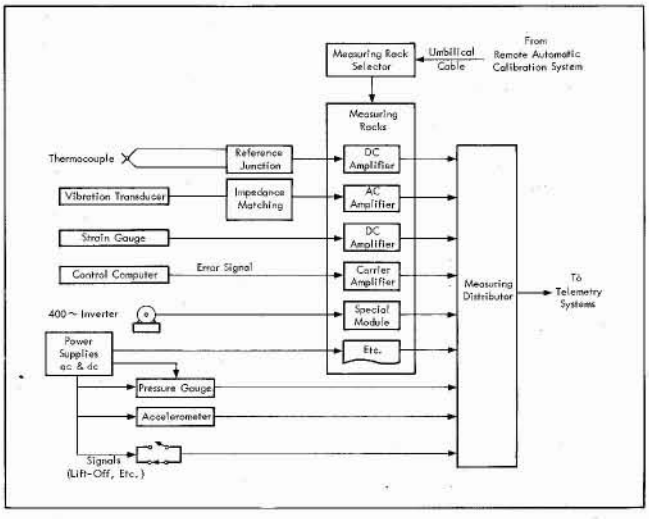
To further increase the amount of data which can be effectively handled, the LVDC is employed to sequence transmission of certain measurements. For instance, once the vehicle has left earth atmosphere, sound levels requiring air for propagation no longer exist, and the LVDC can signal a measuring distributor to switch from an unused or relatively unimportant measurement to those that are more critical for that flight phase. During retrorocket firing, when flame attenuation distorts or destroys telemetry transmissions, signals are automatically recorded by an onboard tape recorder, and retransmitted later.

To effectively monitor vehicle performance it is necessary to determine precise position of the vehicle at any instant and to track its flight path. The RF system of the IU provides this capability, as well as providing communication with the vehicle's guidance and control equipment during flight.

Several tracking systems are used to determine vehicle trajectory during powered ascent and orbital flight. Consolidation of data supplied by different systems provides best possible trajectory information and increases reliability of data.

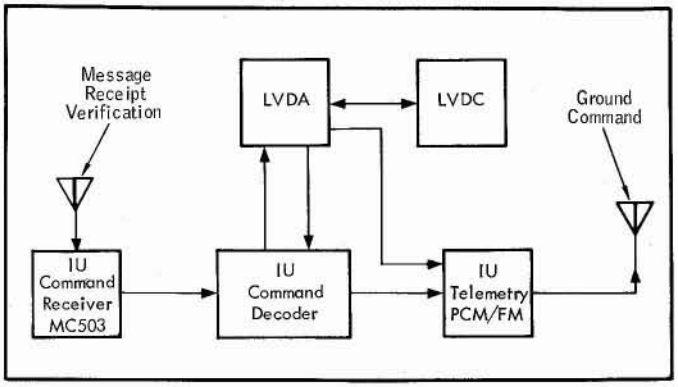
Tracking System

Tracking equipment located within the IU consists of vehicle antennas and transponders, which



IBM DR 2

Typical Saturn Measuring System



IBM DR 7

Saturn IB Instrument Unit Command System

serve to increase range and accuracy of ground-based tracking systems.

A pulse or series of pulses of RF energy transmitted by ground stations in the general direction of vehicle in flight serves to interrogate the airborne transponder. Responding to this query from the ground, the transponder produces a pluse or series of pulses in turn. Triangulation between precisely located ground stations determines point of origin of these reply pulses and fixes location of the vehicle.

Two tracking systems are employed in conjunction with the Saturn IB IU: AZUSA and C-band radar. Two C-band transponders are employed to provide tracking capabilities for this system independent of vehicle attitude. A single transponder is employed with the AZUSA system.

To provide for updating of information in the guidance system, based on real-time evaluation of data obtained from the tracking system or other significant inputs, a radio command link with the vehicle in flight is provided. It is imperative that data transmitted by the ground station is received and interpreted correctly. Any error in updating information could conceivably create a more serious problem than the original information being corrected.

Extensive verification is employed on the ground to ensure that the transmitted message is correct. Similar verification is provided within the IU to ensure that the message received agrees with the message transmitted. The incoming message is accepted only after this verification process is completed.

An incoming message is routed from the system antenna to the command receiver, where it is amplified and demodulated. The message is sent from the receiver to the command decoder for decoding into original pattern of digital bits. First check of message validity occurs here: if

a deviation in bits or a missing bit is detected, the entire message will be rejected. If the command is accepted, it is subjected to a further verification.

First, the vehicle address supplied as part of each command is checked. This address is necessary to differentiate between commands intended for the IU and those intended for use within the vehicle spacecraft—both use similar command links.

If the address is valid, establishing that this message is intended for the IU, the message is then checked to determine intended routing of the command within the guidance system. If no discrepancy is noted, the command is furnished to the LVDA, where it is held for action by the LVDC.

At the same time that the message is released to the LVDA, the command decoder relays a pulse through a telemetry link with the ground station to indicate that a message has been received and processed. A similar pulse is generated by the LVDA to confirm its receipt of the completed message.

Data intended for updating information in the LVDC is stored in the LVDC and read out to telemetry as an additional safeguard against introduction of erroneous information. These bits are sent to the Manned Space Flight Control Center, where they are compared with the message originally transmitted by the ground station. If all bits are verified, an execute command is transmitted. The LVDC will act upon the update data message only upon receipt of this command.

Seven types of messages can be processed: updating LVDC; commands to perform updating; commands to perform tests, special subroutines or special modes of operation; a command to dump or clear certain sectors of the computer memory; and a command to relay a particular address in the computer memory to the ground. Provisions have been made to expand the number of types of messages if experience indicates this is necessary.

Electrical System

The IU electrical system provides power to operate equipment of other systems. In addition, the electrical system contains the emergency detection system (EDS) distributor for detecting abnormal conditions affecting safety of the mission. If severe abnormal conditions are encountered, an automatic abort will be initiated. When less severe abnormality occurs, the electrical system will provide an indication which may be used as a basis for an abort, if deemed necessary.

Similar to other systems of Saturn IB, operation of the electrical system can be divided into two phases: prelaunch and flight. Electrical power during prelaunch is furnished from ground sources through the IU umbilical connection.

At approximately 25 seconds prior to liftoff, a signal originated in the launch control center transfers power to four 28 vdc batteries mounted within the IU. Each battery has a capacity of 300 ampere hours and is capable of operating to an altitude of 15,000 nautical miles above earth. Loads are distributed over the four batteries to equalize drain on each battery and to provide a redundant power source in the event of battery failure.

Two special power supplies are provided: a 5-volt master measuring voltage supply converts 28 vdc main supply to a highly-regulated 5 vdc to be used as a reference and supply voltage for the components of the measuring system; and a 56-volt power supply provides for operation of the guidance and control system's ST-124-M inertial platform.

To provide maximum power utilization with a minimum penalty in battery weight, selective power application is employed throughout vehicle flight. The guidance system's LVDC and LVDA control switching operations, to turnoff unused or unimportant circuits in favor of more important applications as the mission progresses.

Emergency Detection System

The emergency detection equipment monitors thrust for both stages of the vehicle, guidance computer status, angular attack rates, attitude error, and angle of attack. This information is sent to the IU electrical system where it is processed by the emergency detection system (EDS) distributor.

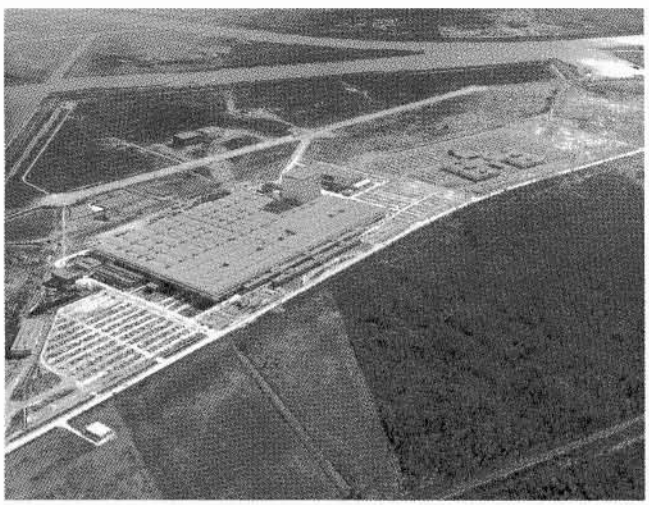
The EDS distributor serves as an interconnecting and switching point, and contains the logic circuits which must decide the nature of an emergency. Serious difficulty allowing no time for human intervention will initiate an automatic mission abort; less urgent problems will result in a visual display in the spacecraft, allowing the astronaut to decide whether an abort is necessary.

FACILITIES

INTRODUCTION

The Saturn IB program largely utilizes facilities that were in existence at the initiation of the program.

With minor modifications, a vast network of shops, laboratories, engineering offices, special test stands, and transporting equipment have supported components, subsystems, systems, and structures that contribute to the entire launch vehicle.

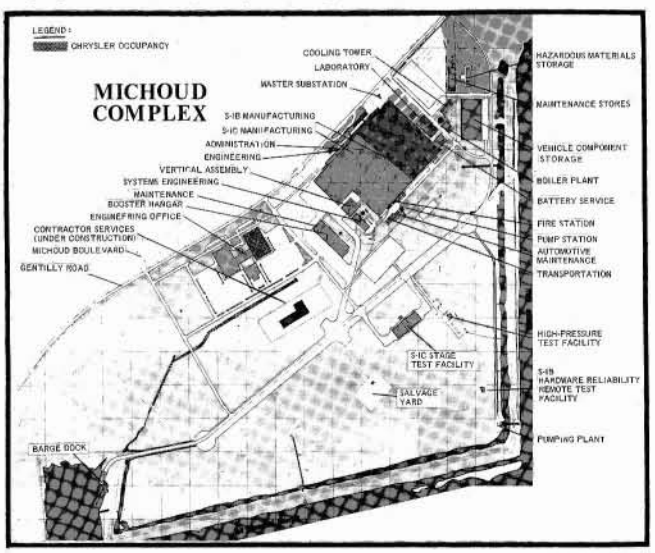


CC-61

Michoud Assembly Facility Aerial View

CHRYSLER FACILITIES

Chrysler Corporation Space Division manufactures the S-IB stage at the 900-acre Government-owned Michoud Assembly Facility located within the New Orleans city limits, approximately 15 miles from the downtown area. The Michoud facility provides 3,188,000 square feet of floor space,



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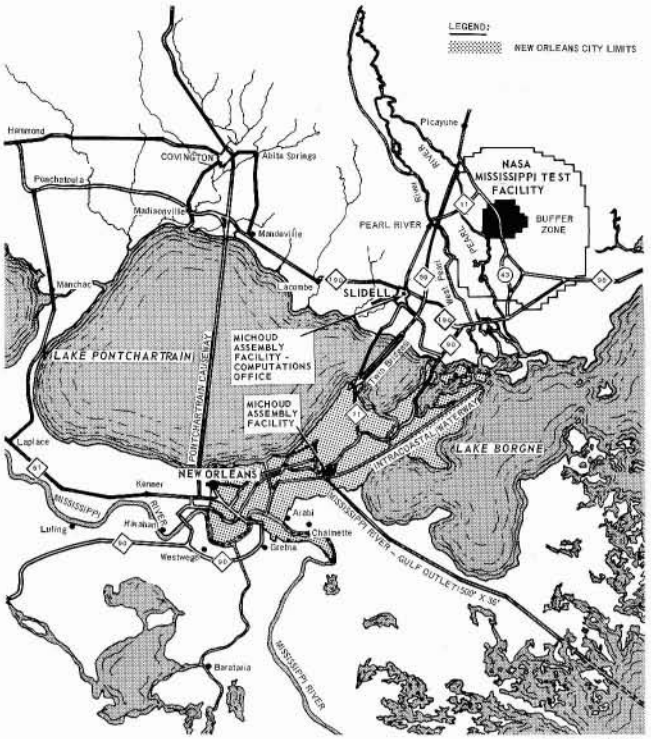
Michoud Assembly Facility Manufacturing and Engineering Complex



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Michoud Assembly Facility Location

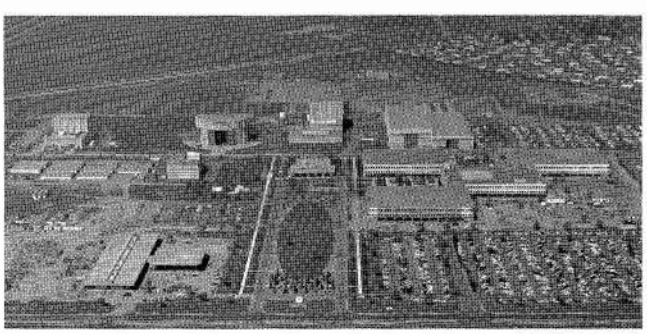
which includes approximately 580,000 square feet of administrative and engineering office floor space and 1,828,000 square feet of manufacturing area floor space. More than 11,500 Government and contractor personnel are now employed at this facility, which is a part of MSFC, Huntsville, Alabama.

Barge Dock

S-IB stages are transported to static firing and launch sites via waterways. The barge dock for the Michoud facility is located 600 yards west of the main plant. A 1,700-foot roadway extends from the west end of an existing runway onto the 75 x 200-foot concrete dock.

DOUGLAS SPACE SYSTEMS CENTER

The Douglas Space Systems Center at Huntington Beach, California, is the master facility for engi-



DAC 441510

Douglas Space Systems Center at Huntington Beach, California

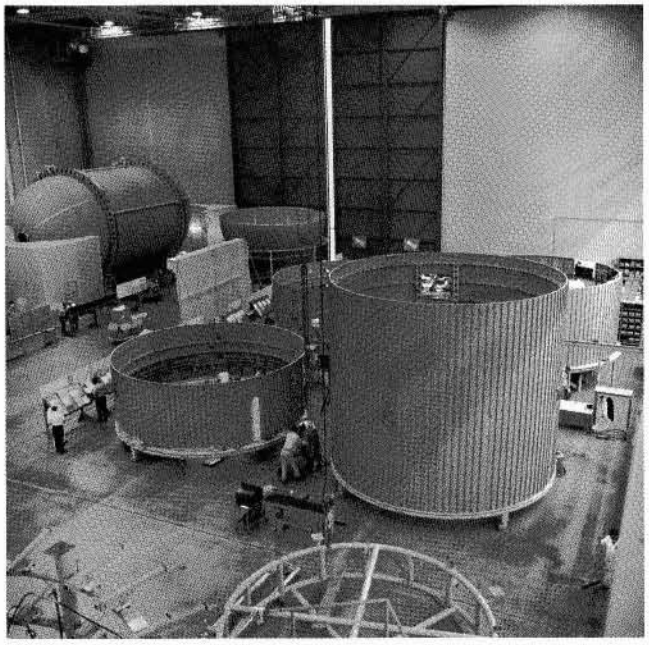
neering and production of the S-IVB stage. Built by Douglas with an investment of company funds which now totals $41 million, the Space Systems Center will grow steadily with the nation's space effort. Subassembly fabrication for S-IVB is accomplished at Santa Monica and Tulsa, Oklahoma facilities and shipped to Huntington Beach for final assembly.

Sacramento Test Center

Major facilities at the 4,000 acre Douglas Sacramento Test Center (STC) that support the S-IVB stage operations include Beta Complex, Gamma Complex, and a Vehicle Checkout Laboratory.

BETA COMPLEX

Beta Complex has twin static firing test stands 1 and 3 for integrated testing of S-IVB stage systems and GSE, and includes propellant and pneumatic storage facilities, test control center, and water and power systems. A complete telemetry system and closed-circuit TV system is also installed on each stand. Test stand 1 was used for battleship stage buildup and testing during the S-IVB stage development program, and has since been converted to support flight stage testing. Test stand 3 is being used for flight stage testing. Design of the Beta Complex permits the eventual addition of the now non-existent test stand 2, if the program requires it. Propellant facilities associated with each test stand include a 175,000-gallon LH_2 storage tank, a 40,000-gallon LOX tank, and control equipment.



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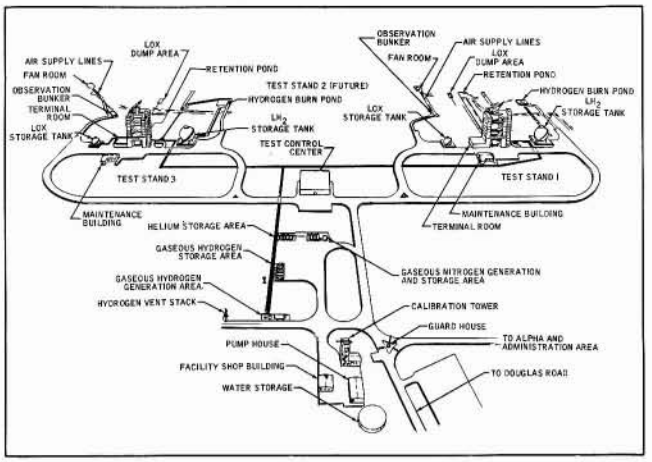
S-IVB Production Assembly and Manufacturing Building

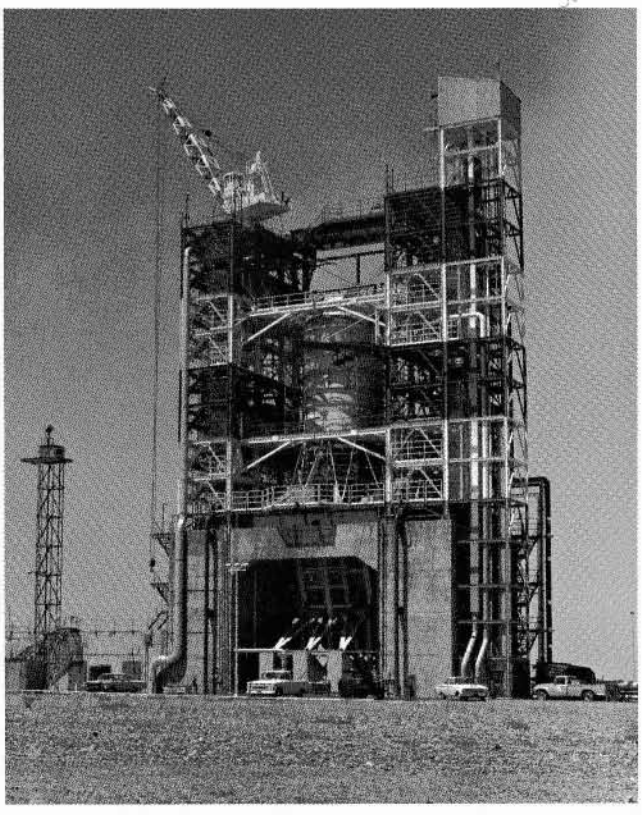


DACO 177A

Beta Complex at Sacramento Test Center

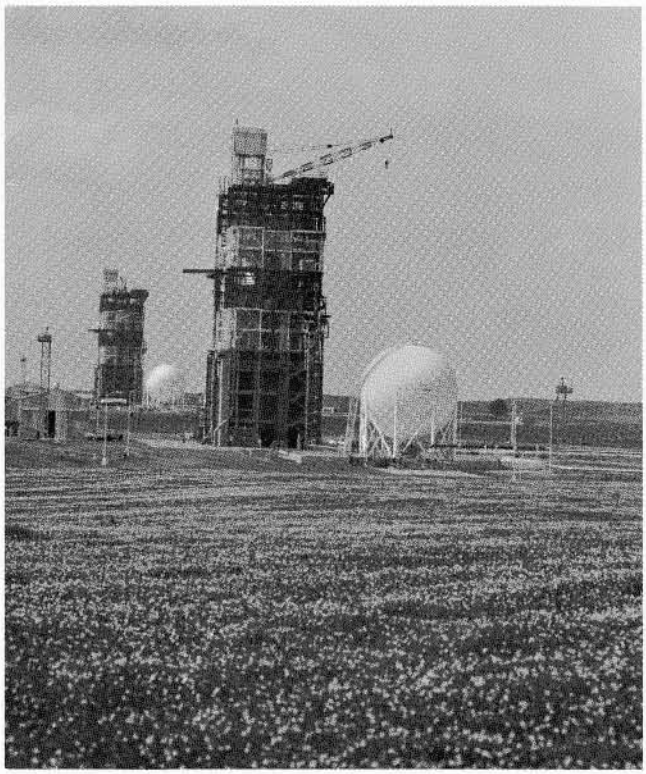
GAMMA COMPLEX

Gamma Complex is a smaller test area for development and testing of auxiliary propulsion system components for the S-IVB stage. The test area includes test cells, equipment area, oxidizer storage facility, fuel storage facility, liquid nitrogen (LN_2) storage area, high-pressure gas storage area, instrumentation center, and test control center.



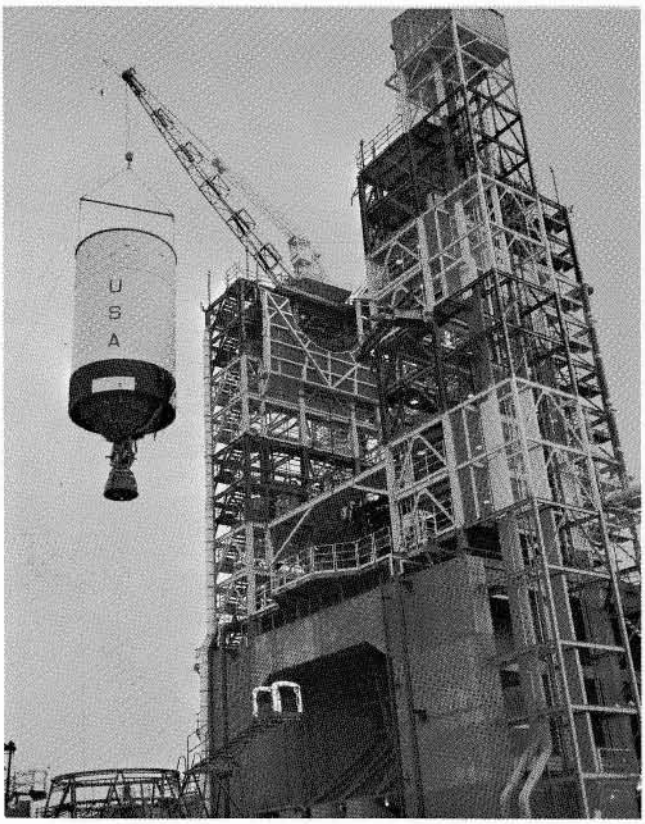
DAC B47-19-26

Static Firing — Test stand at Douglas Sacramento Test Center is one of two built by NASA for static firings of stages prior to delivery to Cape Kennedy. Each flight vehicle undergoes full power, full duration acceptance test, running at full thrust for approximately eight minutes, to prove readiness for launching. In stand is stainless steel battleship stage, used in extensive engine test program.



DAC 16023

Twin Towers — Used for static firings of S-IVB stage at the Douglas Sacramento Test Center, twin towers of Beta test complex are separated by ½ mile, on a 325-acre site. Each tower is 150 feet tall, and designed for up to one million pound thrust loads.



DAC 16637

Douglas S-IVB stage being lifted into Beta Complex test stand.

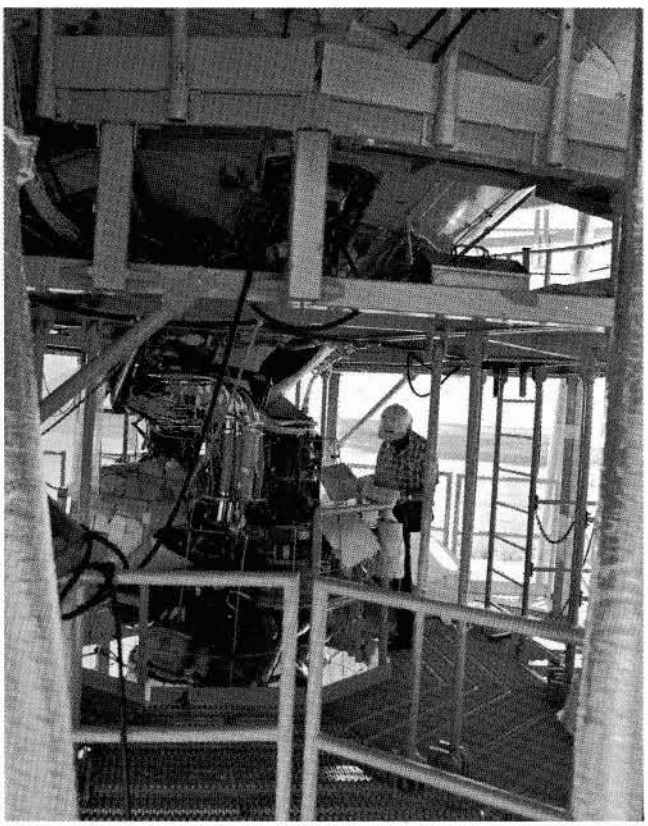
VEHICLE CHECKOUT LABORATORY

The 11-story vehicle checkout laboratory at Sacramento, similar to its counterpart at Huntington Beach, has twin towers more than 115 feet tall for mechanical and electrical systems checkout of S-IVB stages. Automated GSE in the adjoining control center provides the capability for post-firing checkout of stages. In addition to complete subsystem and system checkout and tests, the vehicle checkout laboratory will be utilized for J-2 engine installation and removal, stage modifications, and stage weight and center of gravity determination.

ROCKETDYNE FACILITIES

H-1 and J-2 engines for the Saturn IB launch vehicle are manufactured at Rocketdyne's main complex at Canoga Park, California, and at a plant in Neosho, Missouri. They are tested at the Santa Susana Field Laboratory near Canoga Park. H-1 production test facilities are operated at Neosho.

Manufacturing areas contain general-purpose machine tools for precision and heavy machining, as well as numerically controlled machines, ultrasonic equipment for cleaning purposes, and X-ray inspection equipment. Dust- and temperature-con-

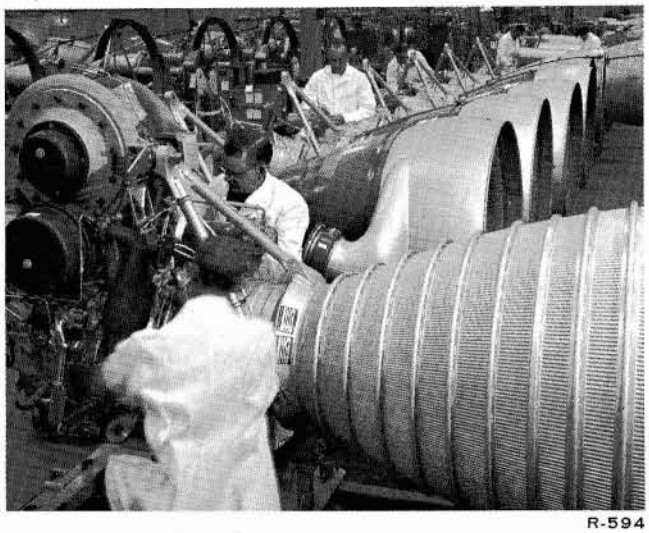


DAC-B47-19-23

Prefiring Check — Test engineer on engine deck of S-IVB test stand at Douglas STC goes through checkoff list for Rocketdyne J-2 engine during preparations for a firing. Catwalks and scaffolding are removed from around engine before test.

trolled white rooms are used for ultra-clean assembly operations and sheet metal preparation.

In addition to the machining and assembly areas, specialized support facilities are utilized in engine fabrication. These include a high-flow water testing facility for checkout of propellant systems, and laboratories for conducting pressure and air calibration checks, vibration, environmental, and structural tests. Thrust chamber tubes are brazed in specially built, gas-fired furnaces.



R-594

H-1 Engine Assembly — Final assembly of H-1 rocket engines is carried out at Rocketdyne, Neosho, Missouri plant. Engines are test fired, then delivered to Chrysler for installation in Saturn IB vehicles.

Seven of the 18 large engine test stands at Santa Susana Field Laboratory (SSFL) are used in development and production testing of H-1 and J-2 engines, including a stand equipped with a steam injection diffusor for altitude simulation testing of the J-2. All of the five Components Test Laboratories at SSFL and one at Neosho are used for static testing of individual components.

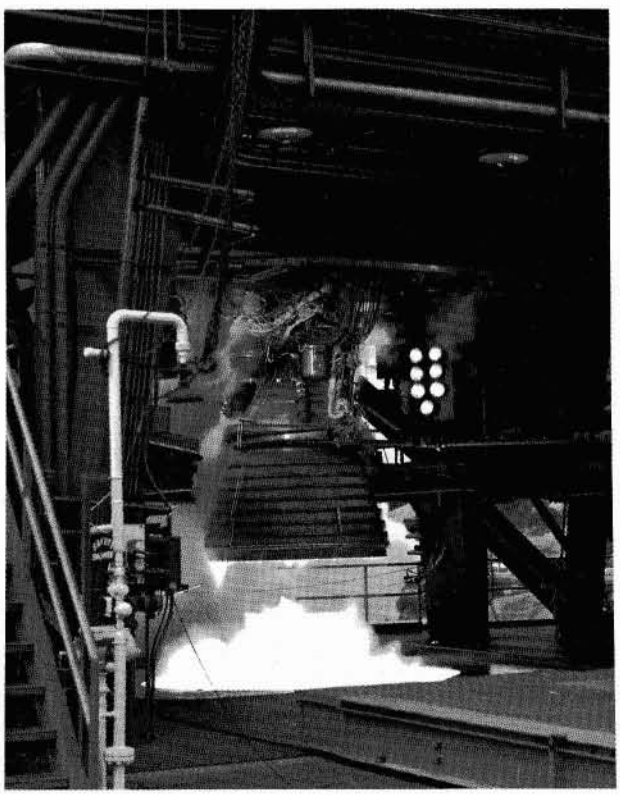
NASA FACILITIES

S-IB Stage Static Testing

NASA facilities at Marshall Space Flight Center, Huntsville, Ala., are used by Chrysler for static and dynamic testing of Saturn IB booster stages. S-IB production stages have undergone hot firings on the static test stand in the MSFC East Area since April, 1965. Stage performance is recorded by instruments located in the area's central control blockhouse. The 178-foot tall, two position, static test stand was built in 1956 for captive testing of the Redstone missile. Since that time, the facility has played a key role in several of the nation's best known rocket programs: Jupiter, Juno II, Saturn

R-TF 584

Altitude Test — Hydrogen fueled J-2 engine is fired under simulated space conditions on altitude test stand at Rocketdyne Santa Susana field laboratory.



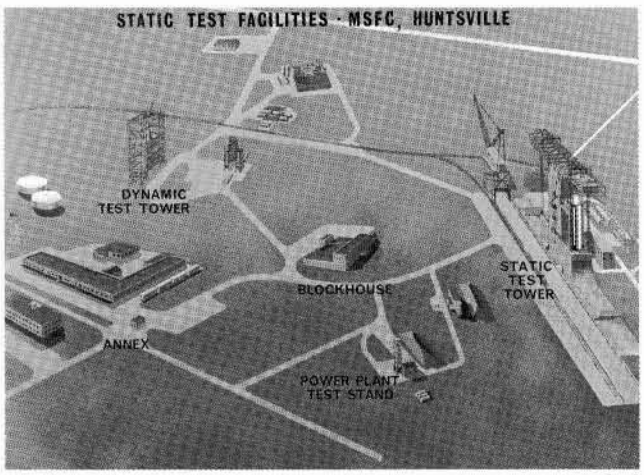
R-ICT24

J-2 Engine Test Firing — Santa Susana Field Laboratory

I, and Saturn IB. Although the test stand has been modified many times in the past nine years, the the basic stand is still much the same. The Redstone missile developed only 75,000 pounds of thrust, less than $\frac{1}{20}$ of the power of systems being tested on the stand today.

Dynamic Testing

Dynamic testing of the Saturn IB launch vehicle is presently being conducted on the MSFC dynamic test stand. The test stand tower is 204 feet high containing approximately 600 tons of structural steel, and includes an electric elevator, a 75-ton stiffleg derrick, and a 50-ton auxiliary hoist. Recently, a complete Saturn IB launch vehicle was placed in the stand for shake tests. Electromechanical shakers moved the big vehicle to determine its vibration and bending characteristics.



CC-13 (D)

Static Test Complex — This illustration shows the location of various engineering and test facilities of the static test complex.

Other Testing Facilities At MSFC

Other related test facilities include a single H-1 engine test stand, a J-2 test stand, a components test facility, and turbopump test positions.

KENNEDY SPACE CENTER LAUNCH SUPPORT FACILITIES

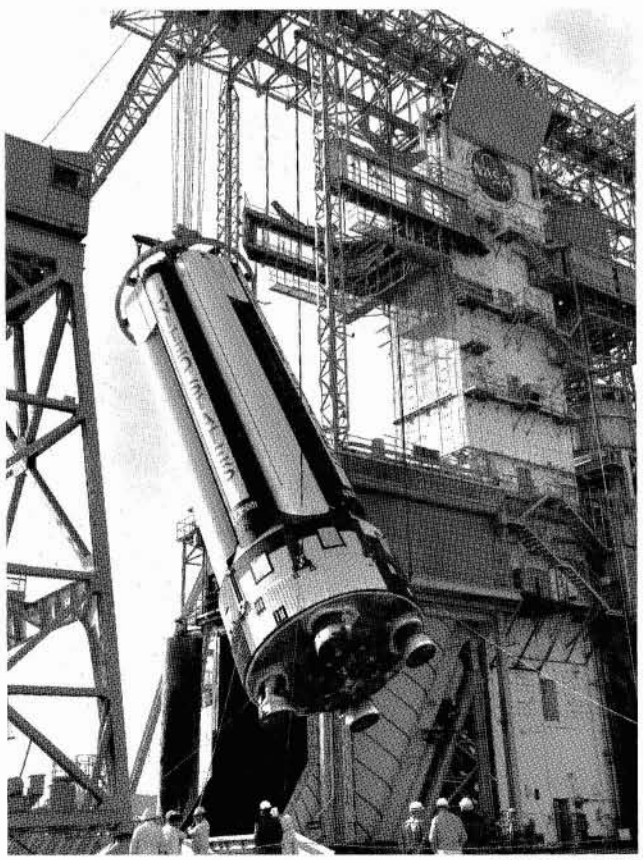
Saturn IB launch vehicles are assembled and checked out at Launch Complexes 34 and 37 at Kennedy Space Center. Available at the launch complexes are primary power source for the payload, environmental conditioning equipment, mechanical handling and transporting equipment, and GSE for testing. GSE for special scientific equipment and payload integration is installed at the launch pad. The Saturn IB is the first large space vehicle whose systems are tested by fully automated GSE (the Automatic Checkout System) controlled by a general purpose computer.

Launch Complex 34

Launch Complex 34 — to be used for the first Saturn IB's — includes a single 430-foot diameter launch pad with a launch pedestal and umbilical tower, automatic ground control station, service structure, launch control center, and operations support building. Service systems at this launch complex include an RP-1 system, liquid hydrogen system, liquid oxygen system, high-pressure storage battery for gaseous nitrogen and helium, high pressure hydrogen battery, and communications system. A converter-compressor facility for gaseous nitrogen and helium serves Launch Complex 34 and Launch Complex 37.

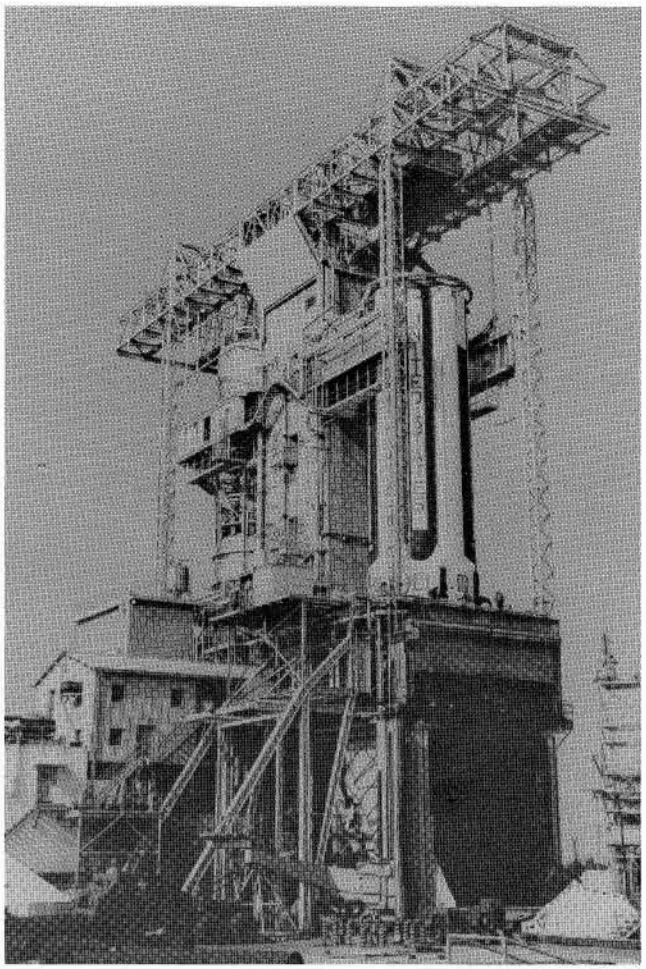
Launch Complex 37

Launch Complex 37 includes two similar 300-foot square launch pads situated at opposite ends of a 1,200-foot long service railway. Each launch pad has a launch pedestal, umbilical tower, and automatic ground control station. One service structure and one launch control center serve both pads. Service systems common to both pads include an RP-1 system, liquid hydrogen system, liquid oxy-



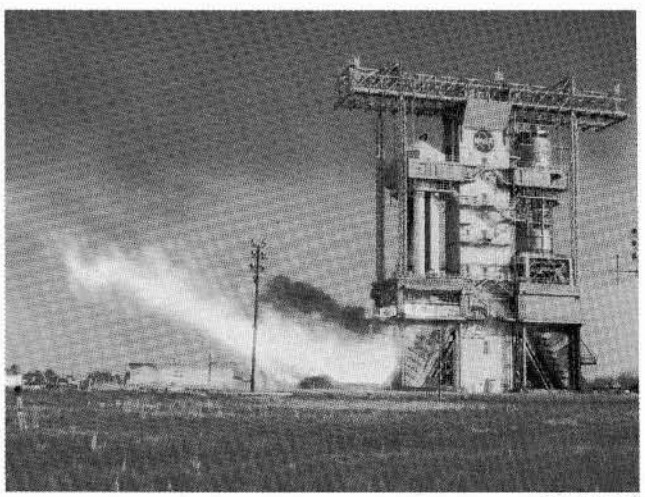
NASA 5-20700

Lifting S-IB Stage — The first S-IB stage is lifted into the static test stand at MSFC.



CC-13 (B)

S-IB Stage Static Test Tower — S-IB stage is readied for static test firing. Annex in left foreground houses engineering offices, instrumentation and control terminal rooms, equipment used in the stage preparation, a facility power system, and a water system for the flame deflector and fire extinguisher system.



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S-IB Stage Static Firing — Flames from H-1 engines spew from the test tower flame deflector. Stage is subjected to a short duration test of approximately 35 seconds and a long duration test of approximately 138 seconds.

gen system, high-pressure storage battery for gaseous nitrogen and helium, high-pressure gaseous hydrogen battery, and communications system.

Launch Complex Facilities

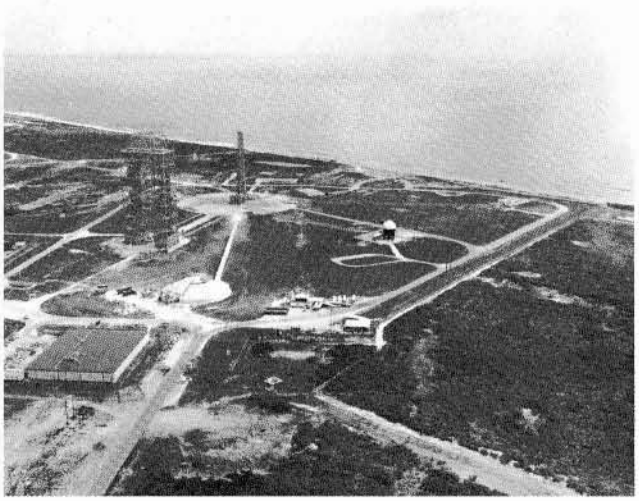
The following facility descriptions apply to either Launch Complex 34 or 37 unless specified otherwise.

LAUNCH PAD

The surface of the launch pad has a maximum height of 16 feet above sea level. The pad area is covered with fire brick to minimize erosion caused by first stage exhaust.

LAUNCH PEDESTAL

The reinforced concrete launch pedestal provides a platform supporting the vehicle and GSE. The



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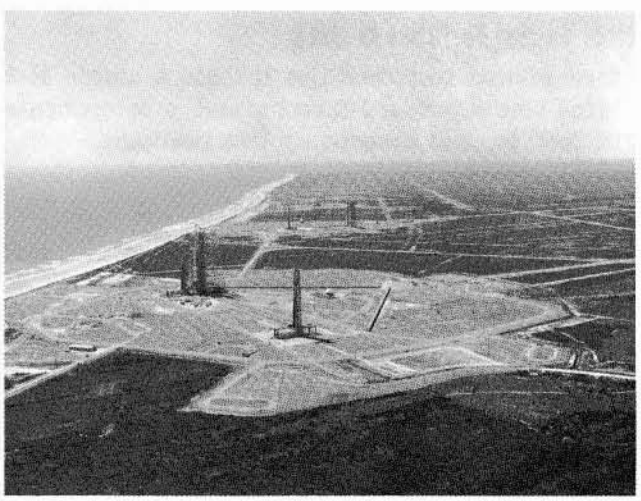
Launch Complex 34 Overall View



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Launch Complex 37 Overall View

platform and leg surfaces exposed to vehicle exhaust are covered with steel plate. An opening in the center of the pedestal allows access to the vehicle boattail section and permits passage of exhaust to the flame deflector located below. Hardlines, ducts, and cables are installed as an integral part of the launch pedestal for RP-1 and LOX systems, water system, pneumatic system, electrical system, and boattail environmental conditioning systems.

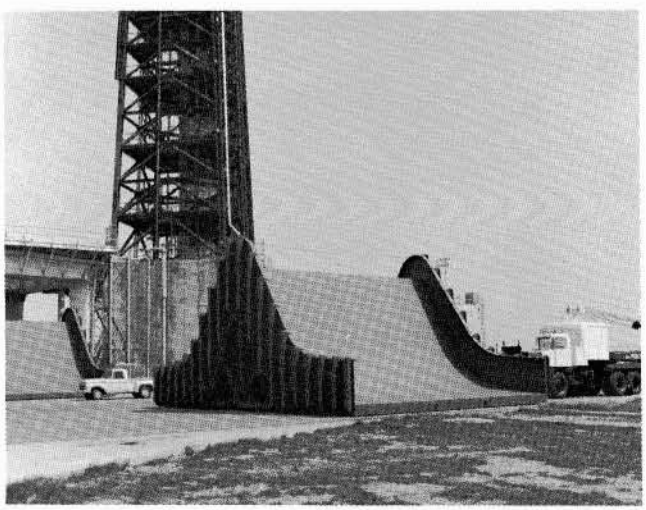
Eight identical holddown arm assemblies are bolted around the opening in the top of the launch pedestal to support the vehicle at each fin. Each assembly consists of a cast steel frame, holddown arm and linkage, ball-lock separator, load cell, and related pneumatic lines and valves. The arm assemblies are connected by a manifold designed to insure simultaneous release. Pneumatic pressure releases the ball-lock separators and frees the holddown arm linkage, permitting the holddown arm to pivot up and away from the vehicle at liftoff.

FLAME DEFLECTOR

The 150-ton flame deflector protects the boattail section of the vehicle, the launch pedestal, and the pad surface from the rocket's flame during ignition and liftoff. The inverted V-shaped, dry flame deflector is constructed with a series of roof-type trusses formed at an 80-degree angle. The trusses are covered with a 1-inch steel skin, which in turn is covered with 4 inches of a special heat-resistant refractory coating.

UMBILICAL TOWER

The umbilical tower and swing arms provide support for hydraulic, pneumatic, cryogenic lines, electrical cables, and environmental conditioning ducts to the vehicle and the spacecraft. The tower is a steel-trussed structure, 240 feet high including the base building (260 feet high at Complex 37). Mounted on the tower is a 5,000-pound capacity boom hoist. The hoist is equipped with a trolley that extends the hook 27 feet from the boom pivot point. The boom can be pivoted 360 degrees.



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Launch Complex 37 Flame Deflector

NASA 100-KSC-65C-3827

Launch Complex 34 Umbilical Tower

Swing arms in the umbilical tower provide support for the interface between the ground supply systems and the launch vehicle during ground operations. Each swing arm is attached to the umbilical tower by a hinged joint, and contains a rotary hydraulic actuator and a hydraulically operated lock pin. At release command, prior to or during liftoff, the swing arms automatically uncouple from the vehicle and swing away.

The Apollo access arm at the 220-foot level (228-foot level at Complex 37) of the umbilical tower consists of a basic arm assembly, an extension assembly, and an environmental chamber bolted together to form an integral assembly. The access arm provides support for service lines from the umbilical tower to the vehicle, and allows astronauts to enter or leave the Apollo spacecraft command module through the environmental chamber.

In case of emergency, the arm can be rotated back to the vehicle in 30 seconds.

The elevator in the umbilical tower, in addition to normal service, provides a fast, safe, and reliable emergency path to safety for astronauts in the event of a decision to delay launch. The elevator is designed to operate in one of two separate modes, egress (emergency) or normal, and is capable of carrying 3,000 pounds at 450 feet per minute. Mode selection is remote controlled from the launch control center.

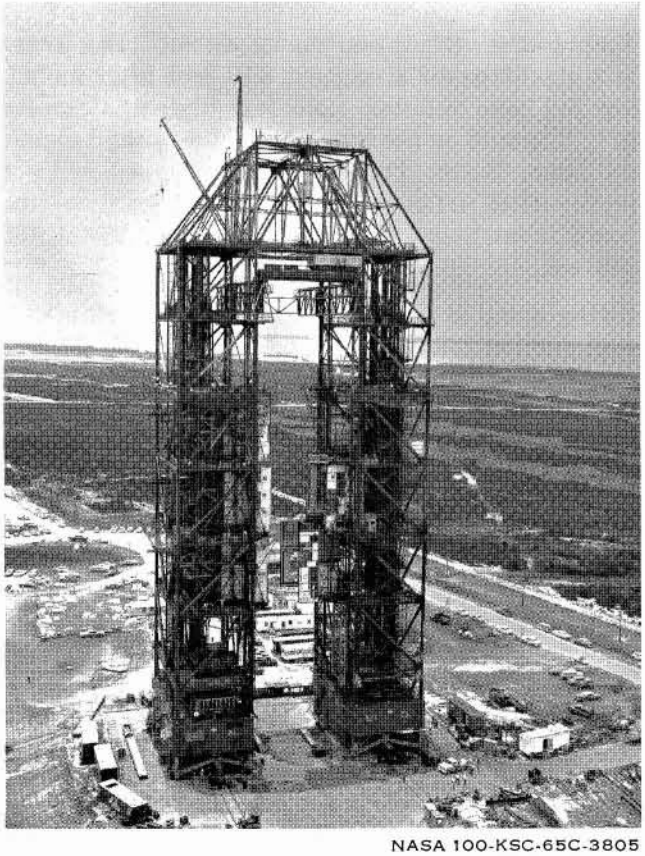
AUTOMATIC GROUND CONTROL STATION

The automatic ground control station is contained in a concrete structure located beneath the umbilical tower. This facility contains digital computers and checkout equipment used during vehicle prelaunch tests; serves as a distribution point for cables; provides space for vehicle test power equipment; and serves as a distribution point for all high pressure gases.

SERVICE STRUCTURE, LAUNCH COMPLEX 34

The 3,500-ton, rail-mounted service structure is 310 feet high and provides for vertical erection, total assembly, weather protection, and checkout of Saturn IB launch vehicle. The overall base dimension is approximately 130 feet by 70 feet. The service structure has a traveling bridge crane with a main 60-ton hoist and an auxiliary 40-ton hoist, with hook heights of 245 feet each and reaches of 28 feet forward and 20 feet lateral. Two traveling hoists, with a travel of 93 feet at the 303-foot, 6-inch level, provide for Apollo spacecraft launch escape system handling and erection.

The service structure is equipped with seven fixed platforms within the structure legs, and eight enclosed retractable platforms. In addition, there are two retractable platforms at the launch escape system hut levels of the structure. Four hinged hurricane doors, 44-foot tall, provide protection to the launch vehicle first stage. Retractable silo sections protect the launch vehicle second stage, instrumentation unit, and the Apollo spacecraft. The tower is also equipped with two elevators in each leg for personnel and freight. The two 25-foot high base sections of the service structure contain space for maintenance and measuring purposes, pneumatic and hydraulic systems, and house a diesel-driven motor generator that drives the four 100-horsepower traction motors which provide self-locomotion of the structure on a dual track railway to or from the launch position. During the final phases of the launch, the service struc-



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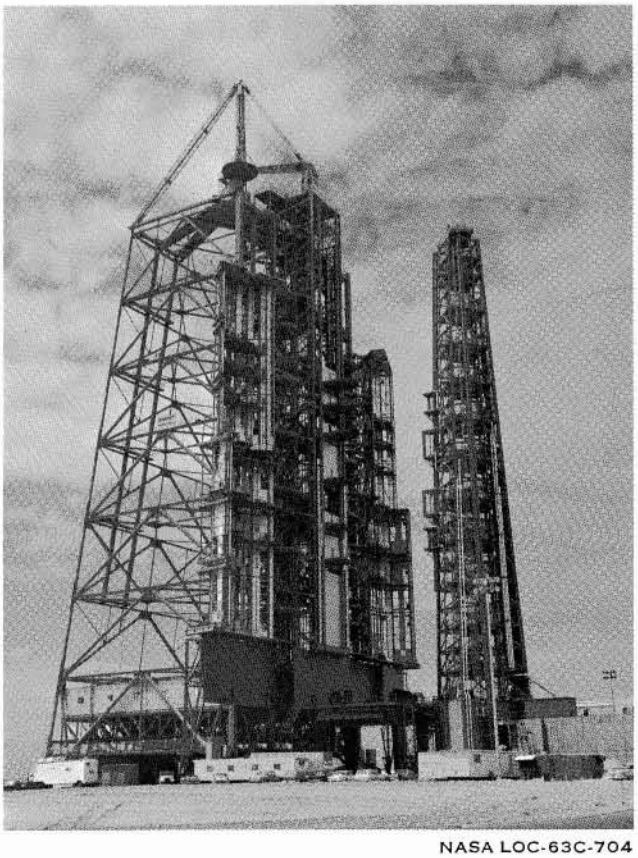
Launch Complex 34 Service Structure



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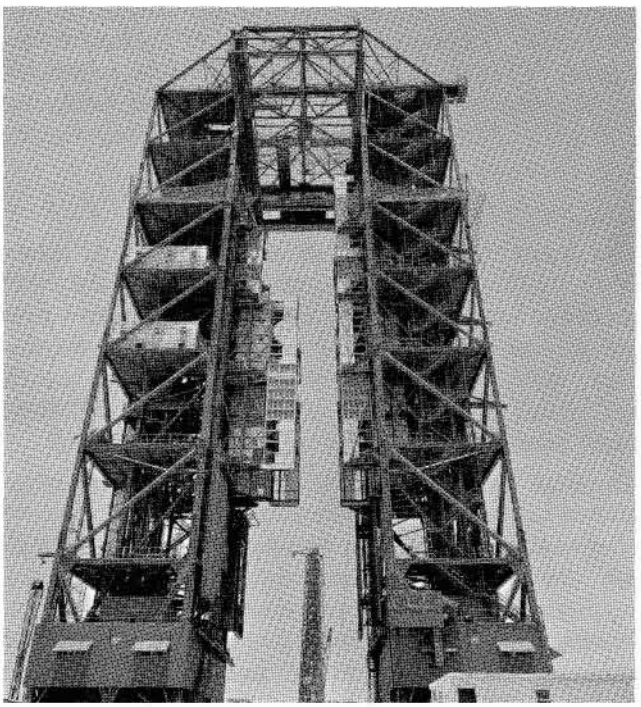
Launch Complex 37 Service Structure and Pad B Umbilical Tower



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Service Structure — The track mounted, wheeled service structure is used to erect the rocket on the launch platform and prepare it for launch. Once launch preparations are complete, the service structure is moved away from the launch platform.

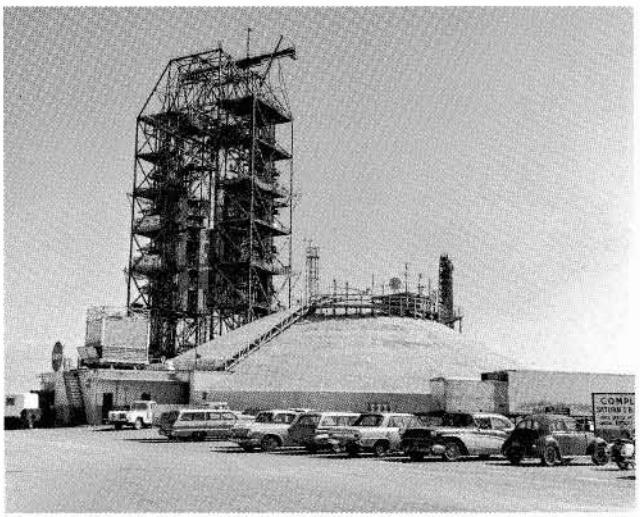
ture is moved to a park position 680 feet from pad center.

The service structure is mounted on four, 12-wheel trucks and is normally anchored to the ground at support points with the load of the structure removed from the wheels. Steel anchor pins secure the structure to the support points when it is in a parked position. When the structure is moved, hydraulic jacks are extended to raise the structure on the wheels. The structure is protected with lightning protection and grounding systems, aircraft warning lights, and illumination for night operations.

SERVICE STRUCTURE, LAUNCH COMPLEX 37

The 5,200-ton service structure provides for vertical erection, total assembly, and checkout of the Saturn IB vehicle. The electric-powered, self-propelled structure is mounted on four trucks that ride on rails, and is driven by four, 100-horsepower, electric motors. The driven side has 64 flanged wheels. The opposite side, which is not driven, incorporates 24 flanged wheels. The trapezoidal structure is of rigid-truss construction and extends to a height of 300 feet. Design permits an increase in height if necessary to accommodate future space vehicles.

Vehicle hoisting and erection is accomplished by



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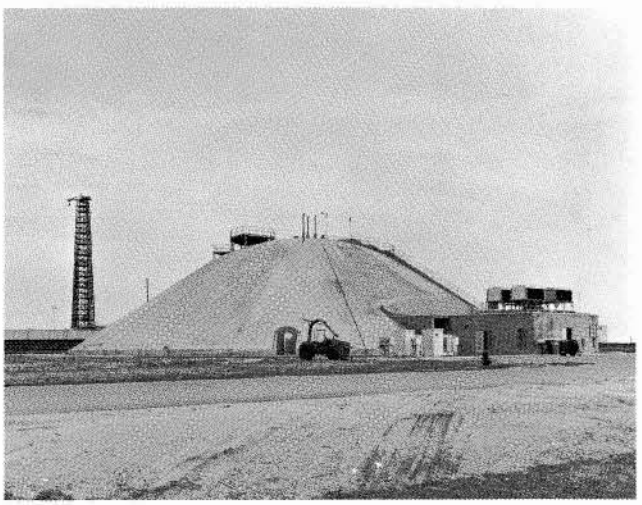
Launch Complex 34 Launch Control Center



NASA-LOC-63-42-9500264

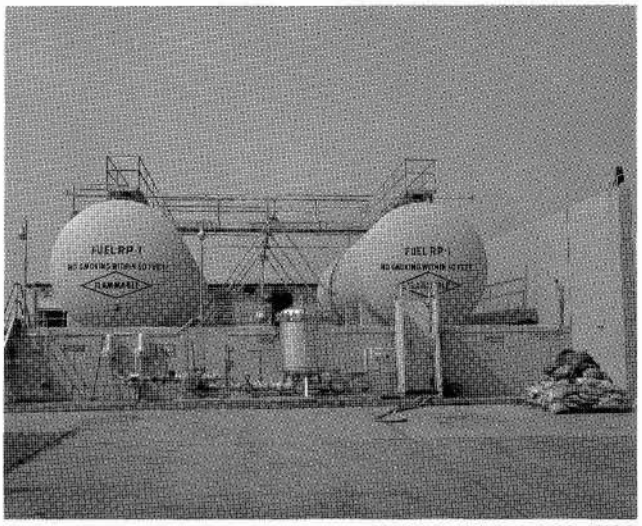
Launch Complex 37 Launch Control Center

means of stiff-leg derrick mounted on top of the service structure. The derrick incorporates a 60-ton hook, 10-ton jib hook, and 40-ton auxiliary hook. The derrick has a capacity of 60 tons, with a 75-foot working radius.

Hoisting machinery is located in the base of the structure. Three elevators, each with a 3000-pound capacity, service all work levels.

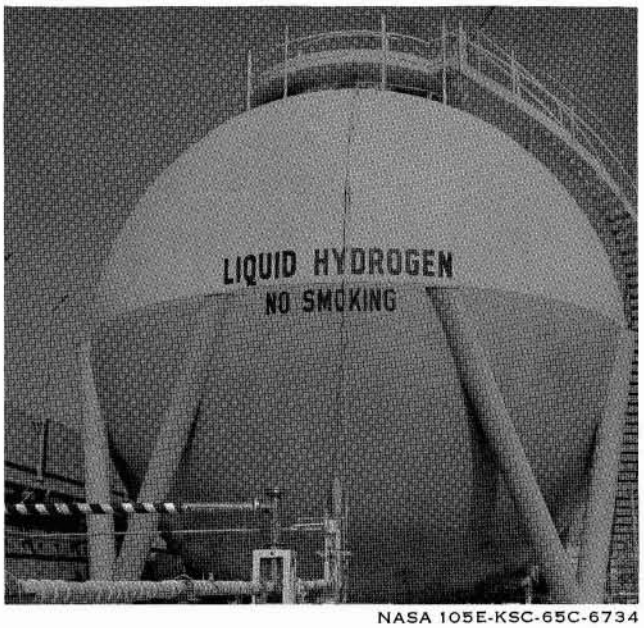
The vehicle and checkout personnel are protected by a hurricane curtain around the launch pedestal from the 35-foot elevation to the 65-foot elevation and six split "silo" enclosures reaching to the 248-foot level. In addition, required service platforms are provided as required by the mission. Hoisting machinery for the platform landings is housed in the base of the structure.

Anchor foundations at each launch pad position



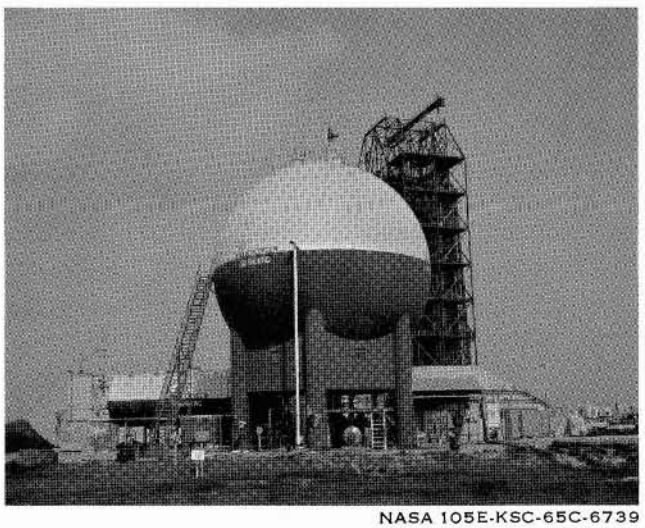
NASA 105E-KSC-65C-6738

Launch Complex 34 RP-1 System



NASA 105E-KSC-65C-6734

Launch Complex 34 Liquid Hydrogen System



NASA 105E-KSC-65C-6739

Launch Complex 34 Liquid Oxygen System

are used to anchor the structure and to remove the load from the wheels. The traction system is designed to propel the service structure in winds up to 35 miles per hour. The structure has lightning and grounding systems, aircraft warning lights, and illumination for night operations.

LAUNCH CONTROL CENTER

The launch control center is a two-story, igloo-type building located approximately 1,000 feet from the launch pad. The building houses and provides blast protection for personnel, instrumentation, and control equipment required for launch activities. The building is designed to withstand blast pressures of 2,188 pounds per square inch.

The first floor of the launch control center is partitioned into enclosed areas and contains the ground checkout computer, communications equipment, air conditioning equipment, RF and telemetry equipment, and instrumentation required for monitoring and tracking operations. The second floor contains the test conductor and test supervisors consoles, monitoring and recording panels, television monitors, and periscopes for visual observation of the launch operation.

RP-1 SYSTEM

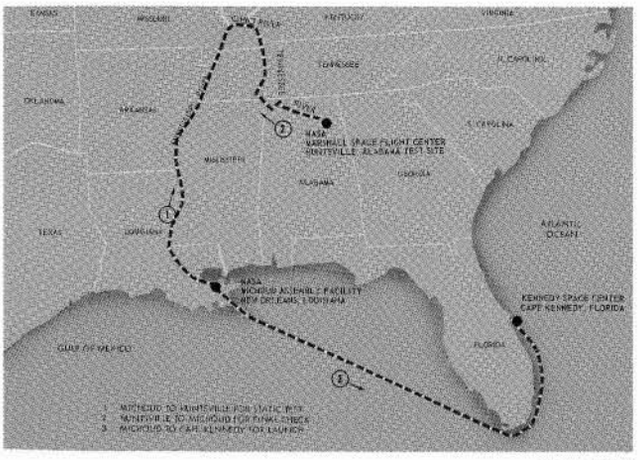
The RP-1 system is used to store and transfer fuel to the S-IB stage. The system is a remote-controlled, automatic/semiautomatic system consisting of equipment located at a RP-1 storage area, automatic ground control station, and the launch control center. Fuel required by the vehicle is supplied from a 43,500 gallon capacity horizontal underground storage tank at Complex 37 and from two horizontal 30,000 gallon capacity tanks, above ground at Complex 34.

OPERATIONS SUPPORT BUILDING

Located adjacent to Complex 34 launch control center, the operations support building provides approximately 30,000 square feet of floor space for use as specialized maintenance areas and for engineering activities at both complexes. During prelaunch operations, the building provides laboratory space for measurement and calibration of telemetry and GSE, electrical networks, and checkout and evaluation of components. In addition, space is provided for critical parts storage, mechanical equipment, and personnel work areas. Space is also provided for prelaunch test and postlaunch data review and evaluation.

LIQUID HYDROGEN SYSTEM

The liquid hydrogen system supplies fuel to the S-IVB stage. The facility is a remote-controlled automatic/semiautomatic system consisting of equipment located at the storage area, automatic


MARSHALL SPACE FLIGHT CENTER
HUNTSVILLE, ALABAMA TEST SITE
MICHOUD ASSEMBLY FACILITY
NEW ORLEANS, LOUISIANA
KENNEDY SPACE CENTER
CAPE KENNEDY, FLORIDA
ATLANTIC OCEAN
GULF OF MEXICO
1. MICHOUD TO HUNTSVILLE FOR STATIC TEST
2. HUNTSVILLE TO MICHOUD FOR FINAL CHECK
3. MICHOUD TO CAPE KENNEDY FOR LAUNCH

CC-12 (C)

S-IB Stage Water Transportation Routes

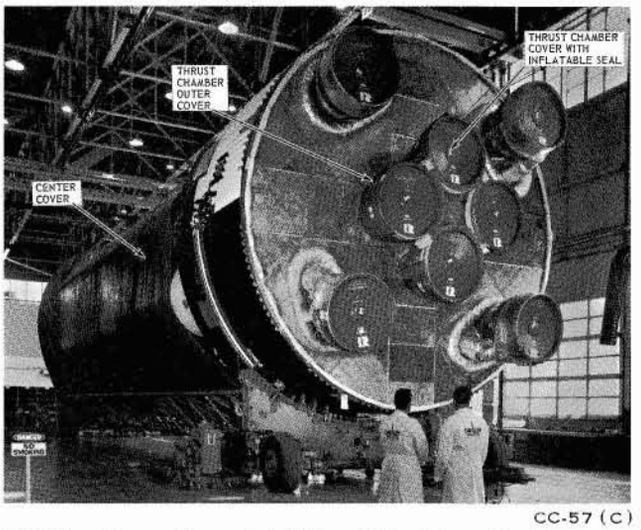
ground control station, electrical equipment house, umbilical tower, and launch control center. The LH_2 storage tank has a capacity of 125,000 gallons.

LIQUID OXYGEN SYSTEM

The LOX service facilities are used to store and transfer LOX to stages of the launch vehicle during fill and replenish operations. Two storage tanks are provided: a 125,000 gallon main storage tank and a 13,000 gallon replenishing tank (28,000 gallon tank at Complex 37).

D-ORM169B

S-IVB Stage Transportation Routes



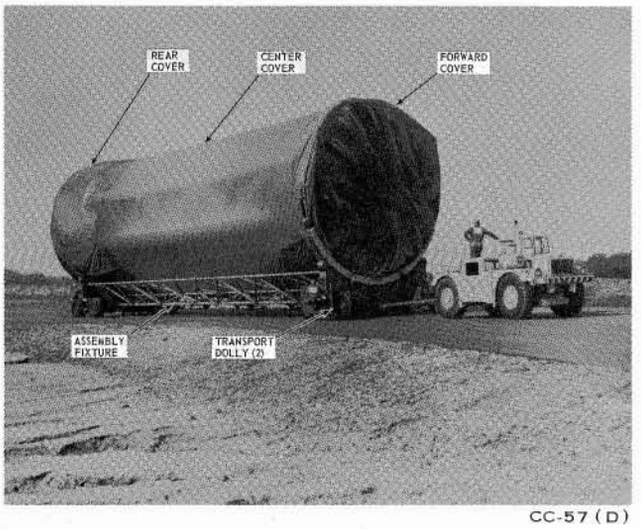
CC-57 (C)

S-IB Stage Preservation — Installation of the tail section cover completes preparation of the S-IB stage for shipment. The eight rocket engines are shown with protective covers and throat seals installed. Desiccant is packed within the throat area of each engine before the throat seals are installed.

COMMUNICATIONS SYSTEMS

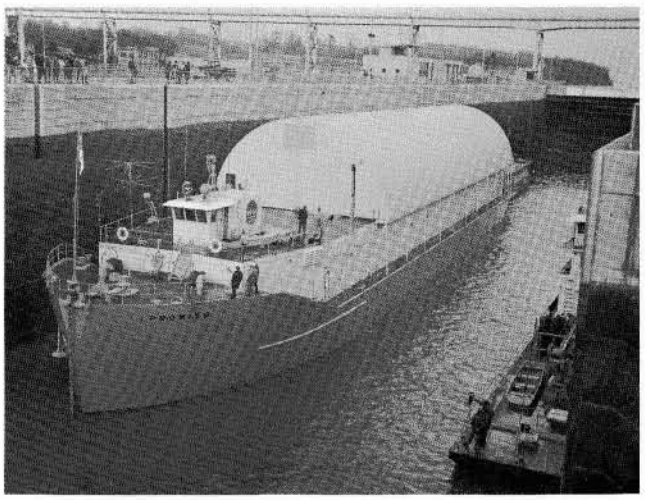
Completely flexible intercommunication, closed-circuit television, timing distribution, and paging systems are provided between all operational areas of each complex. The operational intercom system is a NORCOM, two-wire system compatible with other on-site systems. Various control station panels are tied in with the Eastern Test Range.

Closed-circuit television monitors are located in the launch control center with cameras strategically located to monitor all critical pad operations. These units have a mass switching capability controllable from the launch control center. Numerous views of prelaunch activities in the vicinity of the complex can be selected.



CC-57 (D)

S-IB Stage Preservation Complete — S-IB stage is shown being towed to the barge for shipment. The stage is cradled in the assembly fixture which is supported by two steerable transport dollies.

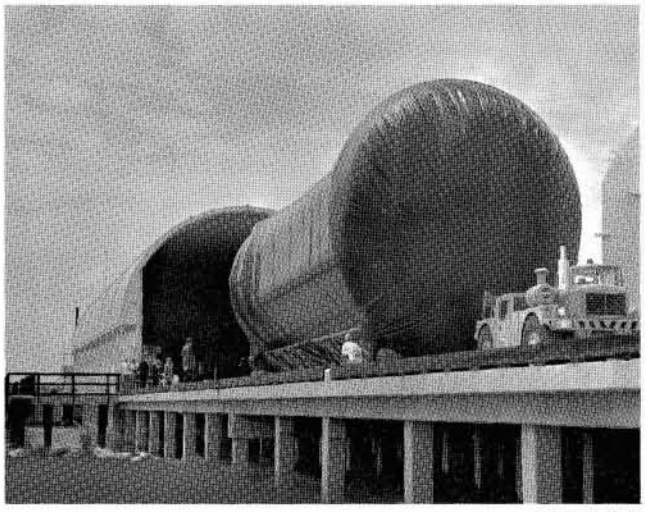


N2-319-3

The Barge Promise — The barge is passing through the lock at Wilson Dam on the Tennessee River.

D-PB-11

Douglas S-IVB Stage and Transporter



CC-12 (B)

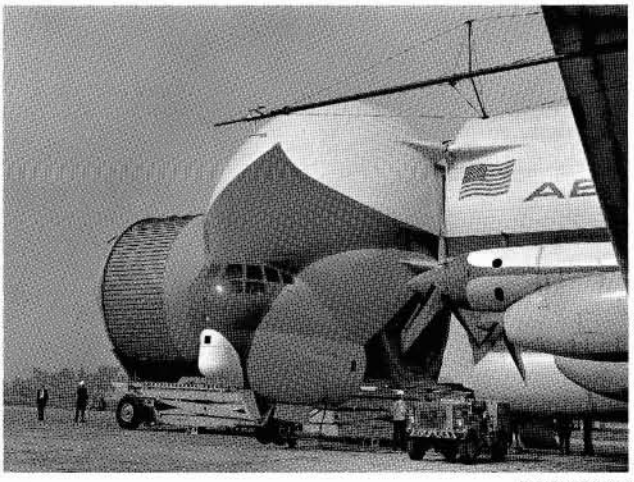
Loading Barge — S-IB stage is loaded aboard the barge Promise.

DAC-16181

S-IVB Journey to Sacramento Test Center

N5-17194

Saturn Barges Arrive — A flotilla of three Saturn barges pushed by a single tug up the Tennessee River just west of NASA-MSFC, Huntsville, Alabama. Aboard the two covered barges are the first and second stages of the Saturn IB dynamics test vehicle, being shipped to Huntsville from New Orleans and Huntington Beach, California for test. The other barge was on its way to MSFC to pick up and return to New Orleans a Saturn V booster weight simulator.



DAC-12022

Loading — S-IVB stage is loaded into Super Guppy.

Distribution of launch countdown timing information is accomplished from the launch control center after generation by range support operations. A paging system is provided and integrated with other systems by a switch system at the test conductor's station.

AIR FORCE EASTERN TEST RANGE TRACKING FACILITIES

The Air Force Eastern Test Range facilities are used for tracking Saturn IB vehicles. The Eastern Test Range (ETR) includes Patrick Air Force Base, Cape Kennedy Air Force Station, and downrange stations with facilities for data acquisition, data processing, data reduction, timing, firing, frequency control and analysis, and range safety systems as well as range wide communications.

The ETR presently extends from the eastern mainland of the United States through the Atlantic area into the Indian Ocean to 90 degrees east latitude. In addition to mainland instrumentation sites, principal tracking stations are located on Grand Bahama Island, Eleuthera Island, San Salvador Island, Grand Turk Island, San Juan (Puerto Rico), Antigua Island, Trinidad Island, Ascension Island, and Pretoria (Republic of South Africa).

TRANSPORTATION AND HANDLING

A unique problem is encountered in transporting and handling S-IB and S-IVB stages. Due to the existing nationwide facility locations for manufacturing, testing, and launching, stages must be moved to fully utilize facility capabilities. Various expeditious methods of transporting and handling stages are being used. Land routes are used for short hauls, with special transporters designed for that purpose. Water routes are used, for the most part, for long hauls with stages loaded on barges. The feasibility of using air transportation became a reality during the Saturn I program when the Pregnant Guppy was used for transporting the Douglas-produced S-IV stage. A larger version, the Super Guppy, is now being flight tested and is planned for transporting S-IVB stages.

D-SGPY

The Super Guppy — Largest aircraft in the world

S-IB Stage Transportation

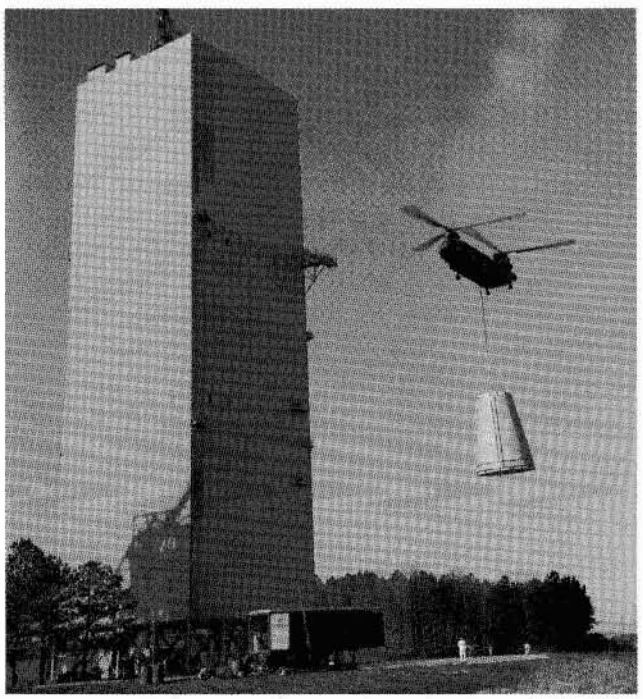
The Chrysler-produced S-IB stage on its transporter assembly is shipped by barge from MSFC-Michoud to MSFC-Huntsville for static test via the Mississippi, Ohio, and Tennessee Rivers. After static test, the stage is returned to MSFC-Michoud for refurbishing and final test and then shipped to KSC-Florida for launch. Since all facilities are accessible by water routes, the stage is shipped on a barge.

TRANSPORTER ASSEMBLY

The fixture used to assemble the S-IB stage is also used as a shipping cradle, which is set on two dolly assemblies, one at each end. A tow bar is then attached to one end and the stage is towed by a prime mover, such as an M-26 military tractor.

Barges

Saturn I barges, Promise and Palaemon, have been in operation more than three years. These pio-



N5-17995

The Big Lift — An Army CH-47A helicopter prepares to unload a Saturn IB adapter, nearly 30 feet long and 22 feet in diameter. The adapter was transported in this way from Tulsa, Okla. to Huntsville, Ala., a distance of 600 miles.

neering vessels continue in use in the Saturn IB program. The smaller of the two, Palaemon, 180 feet long, is mainly used to ferry S-IB stages between MSFC-Michoud Assembly Facility in New Orleans, where they are made by Chrysler, and the MSFC-Huntsville, where they are ground fired.

Promise, 260 feet long, carries the Saturn IB boosters from Huntsville and Michoud to the NASA-Kennedy Space Center, Cape Kennedy, Florida, where they are launched. Promise is used for most Gulf and ocean transportation in this program.

The U.S. Navy has turned over to MSFC five additional sea-going barges of the Promise type. They are known as YFNB class barges. Like the Promise, they are 260 feet long and 48 feet wide. Four of these barges will be used in transporting Saturn V stages and components.

The fifth barge, which had been in storage with the San Diego reserve fleet, was recently modified at a west coast shipyard. It is being used exclusively on the west coast, primarily for movement of S-IVB stages from Huntington Beach, California, where Douglas makes them to the test site at Sacramento. This vessel, known as the Orion, is also being used in a limited number of S-II stage movements from the North American S&ID manufacturing site at Seal Beach, California, to Port Hueneme, a distance of 70 miles, enroute to the site of R&D testing in the nearby Santa Susana Mountains.

The Military Sea Transportation Service (MSTS) operates a seagoing craft for NASA called the Point Barrow. The craft, a modified LSD, is being used early in the Saturn IB program for carrying S-IVB second stages from the west coast. Primary job of the craft is to carry the S-II from Seal Beach to the Mississippi Test Facility. The Point Barrow, now designated an Auxiliary Cargo Dock (AKD) is 465 feet long, 74 feet wide, with a 19 foot maximum draft. Its speed is about 15 knots and it requires about a month to make the round trip through the Panama Canal.

S-IVB Stage Ground Transporter Dolly

The S-IVB stage ground transporter dolly provides support and shock isolation for the S-IVB stage during all normal land, sea, and air transporting operations. The transporter consists of a built up base frame, mounted on a four-wheeled undercarriage at the rear end and two sets of drop legs at the forward end. The undercarriage provides a three-point suspension system, two points at the rear and a third steerable point at the front where it is supported by a towing tractor. Tires of low-pressure and high flexibility characteristics are provided, and air-over-hydraulic disc brakes are installed in all rear wheels. The transporter does not exceed 10 miles per hour on paved roads.

Super Guppy

The Super Guppy, the world's largest airplane, is the second aircraft to be developed by Aero Spacelines, Inc., for air transport of large space hardware. The aircraft has an inside diameter of 25 feet and a total length of 141 feet, 3 inches. Tail height is 46 feet, 5 inches, or almost 5 stories above the ground. Cubic displacement of the fuselage is 49,790 cubic feet, which is approximately five times that of present jet transports. The first flight took place in August, 1965, and a flight test program is under way to qualify the plane to carry the Douglas S-IVB stage, the IU, and the LEM adapter built by North American Aviation. These structures are too large to be transported by any existing aircraft.

Helicopter

A helicopter was pressed into interim service for transporting the conical-shaped adapter which connects the Instrument Unit and the Apollo service module. The adapter is nearly 30 feet tall and a maximum of 22 feet in diameter.

TESTING

TESTING REQUIREMENTS

The philosophy of "all-up" flight tests on Saturn IB launch vehicles requires that all stages and spacecraft modules be complete and fully functional, so that a maximum of data and experience can be obtained from every launch.

Inherent in this approach is the requirement for absolute maximum reliability in the launch vehicle. When engines ignite and holddown mechanisms let go, the vehicle is committed to its mission.

Economics of present large rocket programs require that all possible steps be taken to insure the success of each mission before the countdown begins. This is the reasoning behind the immense ground testing effort in the Saturn program — an effort in which every element, from the smallest component to the complete launch vehicle, is tested repeatedly in comparative safety on earth to prove that it will later work properly in space.

For Saturn IB, each contractor has established a test program that provides a firm foundation for confidence in the launch vehicle. It starts with applied research testing to verify specific principles considered in the basic design of the stage and its components. Every design element, every small part, every major component goes through a similar process: design evaluation testing; qualification testing; production acceptance tests; subsystem testing; system testing; stage and vehicle testing; and finally, flight, which is the ultimate test.

Design Evaluation

Design evaluation testing establishes component configurations which will comply with actual operating requirements, and demonstrates that each part will function under all foreseeable operating conditions.

Qualification Testing

Qualification tests are similar in nature to design evaluation tests, but provide conclusive evidence that the test item will perform as required in environments to which it may be exposed. The program qualifies an article for use on a flight vehicle, and provides test data for documentary proof.

In both design evaluation and qualification testing, each item is actually subjected to a series of several tests in several different environments, namely the conditions of vibration, high-intensity sound, electromagnetic interference, heat, or cold.

For each test, the proper equipment must be provided, a test setup devised that will yield the necessary information, and permanent records provided.

Design evaluation and qualification testing on each Saturn component actually involves dozens of separate test operations in several different environmental laboratories, and may require a period of several months.

Production Acceptance Tests

Production acceptance testing consists of a functional checkout of end-item hardware carried out in a regular program to insure that every article delivered meets operational requirements.

Each rocket engine undergoes three static test firings before delivery to the stage contractor.

All electrical and electronic components undergo a visual inspection and a 100 per cent functional testing. Fluid-carrying mechanical components are subjected to proof pressure and leak tests which exceed normal operating requirements. Structural components are given visual, and often, X-ray inspections and dimensional checks. Explosive components undergo visual and X-ray inspections, and destruction tests on a lot-size basis.

Subsystem Testing

After qualified and accepted parts and component assemblies are brought together in an operating subsystem, further qualification and acceptance-type tests are run on the complete package to demonstrate workability, reliability, and compatibility of the various components.

System Testing

All major systems on Saturn IB stages, and all individual subsystems which can function only after installation on the vehicle, are given final tests in a thorough factory checkout before delivery as a prelude to the final acceptance test procedures.

Final Acceptance Tests

Each completed Saturn IB flight stage is subjected to a series of systems tests which lead to a complete, full-power, full duration static firing which is the final trial before formal acceptance by NASA. On the S-IB, static tests are conducted at the Marshall Space Flight Center, Huntsville, Alabama. S-IVB stage static firings are carried out at Douglas Sacramento Test Center. Following static firing, each stage is put through a detailed post-firing inspection and checkout to determine how

it weathered the stresses of simulated flight and to establish readiness for delivery to the launch site.

Flight Test

Every flight program is designed to provide a mass of information on vehicle performance which is vital to planning future launches. Each stage carries a complete network of instrumentation to measure and record the performance of every system, subsystem, and vital component. Data collected and telemetered back to earth during the few minutes of flight becomes a wealth of information for engineers and scientists which cannot be obtained on earth.

Automatic Checkout

A fully automated, computer-controlled vehicle checkout has been designed into all the major segments of the Saturn IB for extensive stage test operations.

The Automatic Checkout System (ACS) was activated on the S-IVB stage and the IU for the first Saturn IB flight vehicle. For the S-IB stage, it will be activated for the fifth flight stage and all subsequent stages.

Automatic checkout is used first in the final factory checkout. It is brought into play throughout the pre-firing preparations for static tests, and during the actual countdown for the acceptance firings. It is employed again throughout the post-firing checkout, and finally for pre-launch tests at Kennedy Space Center and actual launch countdown.

The ACS uses a carefully detailed computer program and associated electronic equipment to perform the complete countdown of each Saturn stage.

With electronic speed, it moves through a more thorough and reliable checkout test program than is humanly possible. The system performs a point-by-point test of each component; indicates its response to the test, and pin-points any malfunction that occurs. It can also automatically indicate ways to double check a questionable response in order to define any difficulty. It virtually eliminates the possibility of human error during a vital countdown.

Test Documentation

In all Saturn test operations, from design evaluation to flight, documentation of results is as important as the acquisition of data. In order to assure reliability and provide maximum confidence in every vehicle, the performance history of every part, component assembly, subsystem, and system must be accurately detailed and permanently recorded. This formidable task of record-keeping provides a test data bank for Saturn program engineers, and can be an invaluable source of reference in the event of minor or major malfunctions in a test or flight.

VEHICLE ASSEMBLY AND LAUNCH

VEHICLE ASSEMBLY AT KENNEDY SPACE CENTER

Stages and modules that comprise the Saturn IB launch vehicle are brought together for the first time at Kennedy Space Center. There are over 90 operations performed for preflight testing and checkout. The S-IB stage, S-IVB stage, and Instrument Unit (IU) of the launch vehicle, as well as the major Apollo spacecraft modules, undergo numerous, complex testing operations to insure flight-ready space vehicles.

S-IB Stage Operations

The S-IB stage is transported by barge from Marshall Space Flight Center's Michoud plant to the Hangar AF-4 at Kennedy Space Center. After a receiving inspection and pre-erection preparation, the stage is transported to the launch complex and erected on the launch pedestal where the following operations are performed:

1. Installation of Fins
2. Power-on Checks
3. Digital Data Acquisition Checks
4. Radio Frequency and Telemetry Checks
5. Electrical Networks Checks
6. Mechanical Systems Checks
7. RP-1 and Liquid Oxygen Loading Tests

S-IVB Stage Operations

The S-IVB stage is delivered from the Douglas Sacramento Test Center to Kennedy Space Center and transported to the low bay area of the Vehicle Assembly Building for receiving inspection and pre-erection preparation. The following operations are performed:

1. Power-on Checks
2. Digital Data Acquisition Checks
3. Radio Frequency and Telemetry Checks
4. Electrical Networks Checks
5. Mechanical Systems Checks

The S-IVB stage is transported to the launch complex and erected and mated to the S-IB stage on the launch pedestal.

Instrument Unit Operations

The IU is delivered from Huntsville to Kennedy Space Center and transported to a hangar for receiving inspection and alignment of the inertial guidance platform. The IU is transported to the launch pad where it is mated to the S-IVB stage, and the following operations are performed:

1. Cold Plate Checks
2. Power-on Checks
3. Digital Data Acquisition Checks
4. Radio Frequency and Telemetry Checks

Integrated Launch Vehicle Operations

After the S-IB stage, S-IVB stage, and IU have been mated to form an integrated Saturn IB launch vehicle, the following operations are performed:

1. Electrical Mating Checks
2. Switch Selector Function Test
3. Power Transfer Test
4. Propellant Dispersion Functional Test
5. Guidance and Control Tests
6. Exploding Bridge Wire Functional Tests
7. Sequence Malfunction Tests
8. Emergency Detection Tests

Following these tests, the integrated Apollo spacecraft is mated to the Saturn IB Instrument Unit to form the Apollo/Saturn IB space vehicle. The following prelaunch operations are conducted:

1. Integrated Test with Launch Vehicle Simulator
2. Spacecraft/Launch Vehicle Electrical Mating and Interface Tests
3. Spacecraft/Launch Vehicle Malfunction Detection Test
4. Space Vehicle Integrated Test with Umbilicals Connected
5. LES Mate and Thrust Vector Alignment Verification
6. Space Vehicle Radio Frequency Compatibility and Swing Arm Test
7. Space Vehicle Integrated Test with Umbilicals Disconnected
8. Countdown Demonstration Test
   a. Spacecraft Ordnance Installation and Removal
   b. Spacecraft Water System and Oxygen System Servicing
9. Space Vehicle Flight Readiness Test
10. Launch Vehicle Umbilical Connection Verification

11. First Stage (S-IB) RP-1 Loading
12. Second Stage (S-IVB) APS Loading
13. Space Vehicle Launch Countdown

LAUNCH COUNTDOWN

Launch countdown commences many hours prior to the scheduled liftoff. Installation of pyrotechnics, final electro-mechanical checks, pressurization of pneumatic storage spheres, and loading of propellants are some of the terminal events that will be monitored from the blockhouse.

NASA 105B-KSC-66P-186

Saturn S-IB AS 203 is delivered to Launch Complex 37 for extensive checkouts.

LAUNCH VEHICLE FLIGHT EVENTS

The following typical launch vehicle operations are listed for reference to post-liftoff events. The times specified are subject to change and should not be considered final.

Nominal Flight Time (Seconds)	Flight Time Base (Seconds)	Event
0.0	$T_1 + 0.0$	Liftoff; range safety receivers on
10.0	$T_1 + 10.0$	Change from S-IB stage single engine out to multiple engine out capability
39.0	$T_1 + 39.0$	Command S-IB stage tape recorder RECORD
77.6	$T_1 + 77.6$	Time of maximum Q
133.3	$T_1 + 133.3$	Arm S-IB stage fuel and LOX engine cutoff sensors
138.3 (Approx.)	$T_1 + 138.3$ $T_2 + 0.0$	S-IB stage fuel or LOX cutoff sensors actuate
140.3	$T_2 + 2.0$	Command S-IB stage inboard engines shutdown
144.1	$T_2 + 3.8$	Command recoverable cameras ON
146.6	$T_2 + 6.3$	S-IVB stage ullage EBW system primed to fire ullage rockets
147.3	$T_2 + 7.0$	Arm S-IB stage fuel depletion sensors; electrically interconnect outboard engines thrust OK switches
148.3	$T_2 + 8.0$	S-IB stage fuel depletion actuates, or outboard-engine thrust OK switches deactuate, and outboard engines shutdown
149.0	$T_2 + 8.7$	Fire S-IVB stage ullage rockets to settle propellants in tanks
149.1	$T_2 + 8.8$ $T_3 + 0.0$	Command S-IB/S-IVB stage separation; fire S-IB retrorocket to brake spent S-IB stage and S-IVB aft interstage start S-IB stage tape recorder playback and recoverable camera ejection delay timers
150.7	$T_3 + 1.6$	Start S-IVB, J-2 engine after sufficient clearance between stages
155.1	$T_3 + 6.0$	Activate S-IVB stage propellant utilization system
161.4	$T_3 + 12.3$	S-IVB stage ullage rocket jettison EBW system primed to blow off ullage rockets
163.9	$T_3 + 14.8$	Jettison spent ullage rockets to decrease S-IVB stage weight

Nominal Flight Time (Seconds)	Flight Time Base (Seconds)	Event
630.0	$T_3 + 500.0$ $T_4 + 0.0$	J-2 engine cutoff; propellant nearly depleted; engine is stopped on fuel depletion or IU command
	$T_4 + 0.5$ to $+ 2.1$	Auxiliary pump, range safety receiver, and PU system are turned off; all systems, having completed their functions, are turned off to conserve remaining battery power
660.0	$T_4 + 29.1$	Telemetry turned off

PROGRAM MANAGEMENT

NASA ORGANIZATION

NASA has 12 field centers. Three of these, Marshall Space Flight Center, Manned Spacecraft Center, and Kennedy Space Center, are directly responsible for carrying out the Apollo/Saturn program, including Saturn launch vehicle development, Apollo spacecraft development, and launchings.

The Marshall Center and its two sister centers report directly to the Office of Manned Space Flight in NASA Headquarters. The Office of Manned Space Flight joins two other major offices—Office of Advanced Research and Technology, and Office of Space Sciences and Applications — in carrying out the civilian space agency's broad space exploration program. At the top of the organizational structure is the Office of the Administrator for NASA, who is directly responsible to the President of the United States.

MSFC Project Management Organization

The Marshall Center's portion of the Apollo program requires the efforts of more than 6,000 civil service and many thousand contractor employes. Saturn industrial activities are principally concentrated in three areas of the United States:

- the Northeast, with its grouping of electronic industries
- the Gulf Coast Region, for design, production, test, and launch operations
- the West Coast, with its concentration of aerospace industries for design, production, and test work.

In addition, various research projects by scientific institutions and subcontractor production efforts contributing to the Saturn program are spread throughout the nation.

The geographic dispersion of the Saturn effort requires excellent management communication. The Marshall Center must be aware of related programs carried out by other NASA centers—especially the Manned Spacecraft Center at Houston, managing the Apollo spacecraft program, and KSC, responsible for Apollo/Saturn launches.

The Marshall Center's budget for the present fiscal year is about $1.2 billion. Marshall Center must have a well staffed organization responsive to the many changes which can take place in a program of this magnitude.

Program management for the Saturn IB launch vehicle is vested in the program offices of Industrial Operations — one of the two major divisions reporting directly to the center director. The project managers in Industrial Operations monitors the project effort, plans, and budgets. For technical solutions of vehicle problems, the manager looks for assistance to the laboratories of the Research and Development (R&D) Operations divisions.

Marshall Center's laboratories are functionally-oriented in such primary disciplines as mechanical engineering, electronics, and flight mechanics. Collectively, the laboratories provide the deep-rooted technological foundation on which the success of all Marshall projects depends. In the project offices, technical decisions are made cutting across and affecting many areas. To formulate these decisions, they draw upon the full technical resources of the laboratories, which maintain a high level of professional competence. Laboratory personnel work on projects selected to keep their technical knowledge updated and their technical competence at a high pitch. This is the Marshall work bench philosopy — the "dirty hands" approach.

In addition, Marshall uses a unique technical management technique to solve specific technical problems. This tool is the working group, which serves as an integral part of Saturn management. Working groups provide the project manager with specialized technical knowledge and depth in engineering judgment.

There are 13 Saturn working groups, one for each major technical discipline and interface area. Each working group is chaired by a Marshall engineer, and has top level representatives from appropriate R&D laboratories of MSFC, project offices of Industrial Operations, other centers, and contractors. Through the working groups, Marshall is able to concentrate a wide variety of skills and specialized knowledge in a regulated effort between Government and industry.

Working groups attempt to solve technical problems, but are not responsible for the implementing actions. Their recommendations are forwarded to the project manager, who may accept the recommendation and instruct the contractor to proceed. The project manager may be so restricted by the budget or schedule that he must delay the recommendation. Directive actions resulting from working group decisions come as no surprise to the prime contractors, because these actions are the result of organized thinking and coordination in which the contractor has participated.

Marshall Center instructions to the prime contractor are transmitted through the MSFC resident manager at the contractor's plant. The resident manager is the Marshall Center's man in the field. He is the project manager's onsite representative. The resident manager's office is an essential structure element in MSFC project management organization, through which the project office manages contractor operations. Through the resident manager, MSFC maintains a direct contact with contractor operations and is kept informed of the status of all significant program events.

Working in close conjunction with most of the resident managers are MSFC laboratory technical personnel. They are assigned to the resident manager's office to provide him with assistance in resolving technical problems, and to keep the MSFC technical laboratories directly informed of field technical effort. Laboratory participation is dictated by need as determined by project management.

Many people are involved in attaining the final goal. Project management, technical, and contractor personnel are all tied in a close knit group capable of managing this country's large launch vehicle program.

MANNED AWARENESS PROGRAM

Since the inception of the Saturn program, all NASA, prime contractor, and associated contractor personnel have been alerted to the premise that the launch vehicles they build are to be used for transporting human beings in space explorations. This awareness is defined as the Man-Rating Reliability Program, wherein the integration of crew capabilities, mission requirements, and design characteristics of the launch vehicle achieve inherent reliability which is necessary for mission success and crew safety.

To emphasize the importance of the Man-Rating Reliability Program, all prime contractors have initiated Manned Awareness Programs wherein all personnel strive for individual excellence and integrity of work in all phases of design and production. The Douglas VIP theme stresses Value In Performance through Very Important People; PRIDE is the Rocketdyne program for Personal Responsibility In Daily Effort; and CARE is Chrysler Always Requires Excellence. All Manned Awareness Programs encourage cost savings and reliability, as well as pride in workmanship as contributions to this nation's vital defense and space exploration programs.

MANAGEMENT PERSONNEL

NASA

Dr. George E. Mueller, NASA Associate Administrator for Manned Space Flight.

Lt. Gen. Samuel C. Phillips, Director, Apollo Program. Responsible for all aspects of Apollo lunar exploration program, including Saturn launch vehicles and Apollo spacecraft.

Dr. Wernher von Braun, Director, Marshall Space Flight Center, NASA, Huntsville, Alabama.

Dr. Kurt H. Debus, Director, John F. Kennedy Space Center, NASA.

Miles Ross, Deputy Director, Center Operations, John F. Kennedy Space Center, NASA.

MANAGEMENT PERSONNEL

NASA

R. A. Petrone, Director, Launch Operations, John F. Kennedy Space Center, NASA.

Raymond L. Clark, Director, Technical Support, John F. Kennedy Space Center, NASA.

G. Merritt Preston, Director, Design Engineering, John F. Kennedy Space Center, NASA.

Major General Edmund F. O'Connor, Director, Industrial Operations — George C. Marshall Space Flight Center, Huntsville, Alabama. Management responsibilities for all Saturn Programs including the Saturn I, Saturn IB, and Saturn V launch vehicles.

William Teir, Program Manager, Saturn I and Saturn IB projects at George C. Marshall Space Flight Center, Huntsville, Alabama.

MANAGEMENT PERSONNEL

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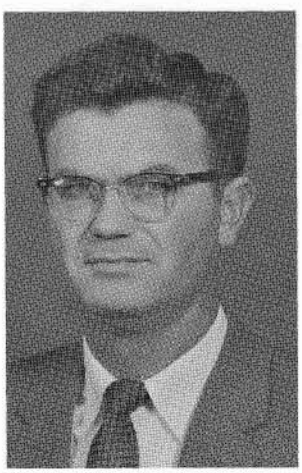
James C. McCullough, S-IVB Stage Manager, Saturn IB Project Office, NASA-Marshall Space Flight Center, Huntsville, Alabama.

William K. Simmons, Jr., Instrument Unit Manager, Saturn IB Program Office, Industrial Operations, Marshall Space Flight Center, Huntsville, Alabama.



Arthur W. Thompson, Chief, S-IB Stage Project, Industrial Operations, Marshall Space Flight Center, Huntsville, Alabama.

MANAGEMENT PERSONNEL

Chrysler

William S Blakeslee, Vice President and Group Executive, Defense-Space, Chrysler Corporation.

H. D. Lowrey, President, Space Division, Chrysler Corporation.

V. J. Vehko, Director of Engineering, Space Division, Chrysler Corporation.

A. R. Trahern, Director of Operations, Space Division, Chrysler Corporation. Responsible for S-IB stage project manufacturing, material, and quality control.

R. F. Gompertz, Director of Systems Test, Space Division, Chrysler Corporation. Responsible for transportation, static test, and launch of S-IB stage.

MANAGEMENT PERSONNEL

MCDONNELL DOUGLAS

Charles R. Able, Chairman and Chief Executive Officer, McDonnell Douglas Astronautics Company.

Jack L. Bromberg, Vice President, Deputy General Manager, McDonnell Douglas Astronautics Company — Western Division.

Theodore D. Smith, Director, Huntington Beach Development Engineering, McDonnell Douglas Astronautics Company — Western Division.

Steven D. Truhan, Director, Florida Test Center for McDonnell Douglas Astronatics Company — Western Division. Directs and coordinates all Company activities at Kennedy Space Center.

Harold E. Bauer, Director, Saturn/Apollo Programs, McDonnell Douglas Astronautics Company — Western Division. Responsibility for all aspects of the present development program on the S-IVB upper stages of the Saturn IB and Saturn V launch vehicles.

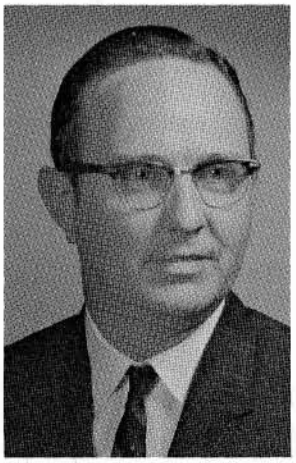
A. P. O'Neal, Director, Saturn Development Engineering, McDonnell Douglas Astronautics Company — Western Division.

MANAGEMENT PERSONNEL

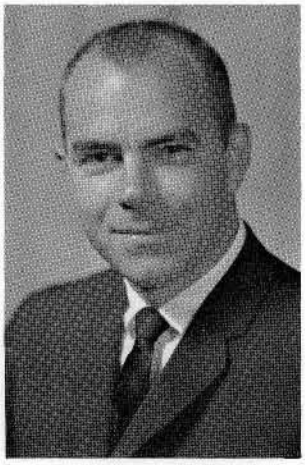
IBM

Bob O. Evans, President of the Federal Systems Division, IBM.

Arthur E. Cooper, Federal Systems Division Vice President and General Manager, IBM Space Systems Center, Huntsville, Alabama.



Clinton H. Grace, Facility Manager, IBM Space Systems Center, Huntsville, Alabama.



Ammon G. Belleman, Facility Manager, IBM Space Systems Center, Cape Kennedy, Florida.

MANAGEMENT PERSONNEL

Rocketdyne

Samuel K. Hoffman, President of Rocketdyne, and Vice President of North American Rockwell Corporation. Directs all Rocketdyne operations for design, development, and production of rocket propulsion systems.

W. J. Brennan, Vice President and General Manager, Liquid Rocket Division, Rocketdyne. Responsible for direction of H-1 engine and J-2 engine development.

Norman C. Reuel, Assistant General Manager, Liquid Rocket Division, Rocketdyne. Assists General Manager in direction of H-1 engine and J-2 engine development.

P.D. Castenholz, Program Manager, J-2 Liquid Rocket Division, Rocketdyne. Directs activities involving development and production of J-2 engine.

N. Dale Johnson, Jr., Program Manager, H-1 Engine Systems, Rocketdyne. Responsible for conduct of all H-1 engine program business and development.

FLIGHT HISTORY

SATURN IB AS-201

The first Apollo/Saturn IB (AS-201) flight followed in the tradition of the completely successful 10 vehicle Saturn I series.

The unmanned vehicle was launched at 11:12 a.m. (EST) on Feb. 26, 1966, from the NASA-Kennedy Space Center's Launch Complex 34.

This launch marked the first flight tests of a powered Apollo spacecraft, a S-IVB stage, and a J-2 engine. The two-stage vehicle achieved all test objectives in its 32 minute suborbital flight down the Atlantic Missile Range.

The mission of the AS-201 flight was to test the launch vehicle and Apollo spacecraft systems.

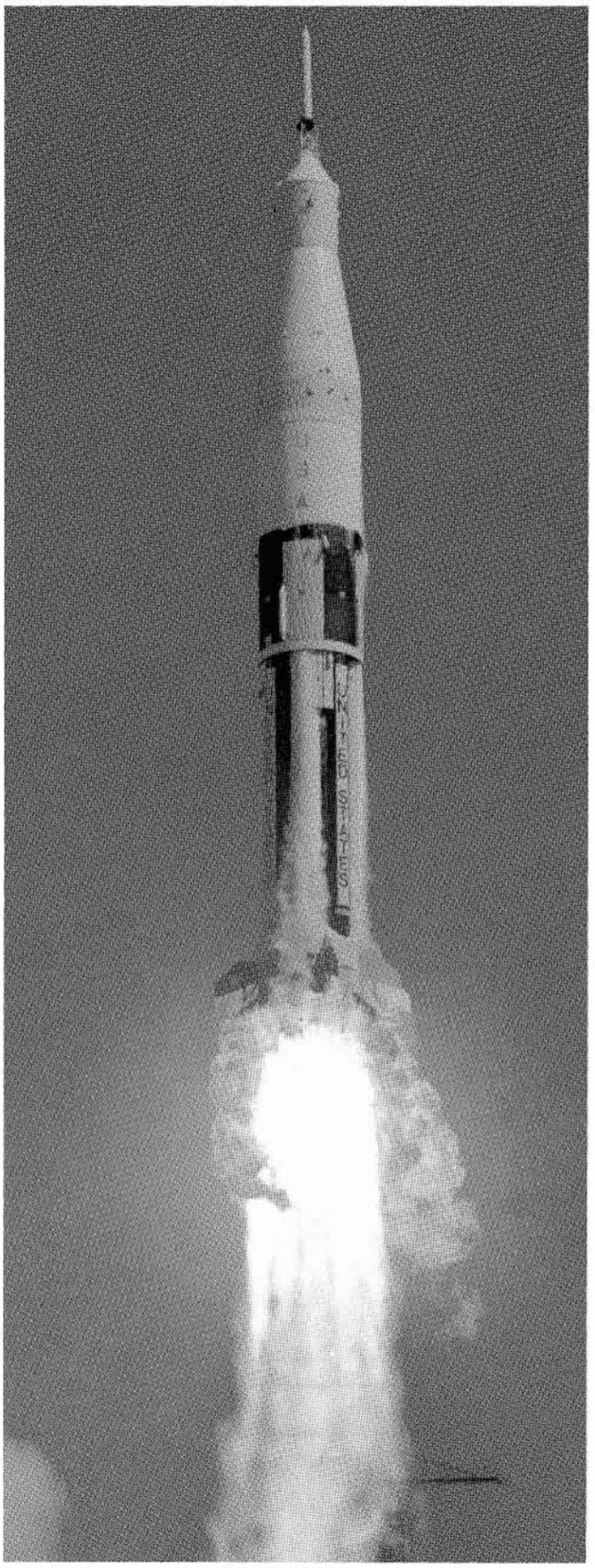
The AS-201 launch was delayed three days because of unacceptable weather in the launch area. On launch date a series of technical problems delayed the firing about three hours. The major trouble was a lack of required pressure in the control pressure system gaseous nitrogen sphere in the booster stage. Nitrogen from this container is used to operate valves, purge certain components, and provide pressurization for engine turbine gearboxes.

Technicians attempted to correct the low-pressure problem, but it persisted — a pressure of 3,000 pounds at ignition was desired. A test was run which determined that by increasing the supply pressure in the ground system the problem would not affect the flight. A decision then was made in the blockhouse to launch the vehicle. The decision was sound, for the proper pressure was maintained throughout the flight.

The Apollo spacecraft, which reached a peak altitude of 306 miles, splashed down about 200 miles southeast of Ascension Island in the South Atlantic.

Performance of the first stage was normal. The four inboard engines cutoff at 141.4 seconds after liftoff, about 0.9 second later than predicted. Outboard engine cutoff came at 146.9 seconds, about 0.4 second later than expected.

The second (S-IVB) stage ignited on command at 149.3 seconds, 0.4 second later than predicted, and was cut off at the desired velocity at 602.9 seconds, burning 10 seconds longer than programmed. The longer burn time was the result of action by the stage propellant utilization system at 240.5 seconds to insure the simultaneous depletion of



DAC-19898

Liftoff — First Saturn IB launch opens a new chapter in the history of the highly successful Saturn family.

SUMMARY OF SATURN FLIGHT PROGRAM

SUBORBITAL FLIGHTS

	Date	Duration	Altitude	Distance	Burn Duration	Remarks
SATURN I						
SA-1	10-27-61	408 sec.	85 mi.	207 mi.	116 sec.	Successful ballistic flight
SA-2	4-26-62	162 sec.	65 mi.	50 mi.	117 sec.	Project High Water I. 96 tons of water exploded.
SA-3	11-16-62	292 sec.	104 mi.	131 mi.	149 sec.	Project High Water II. 95 tons of water exploded.
SA-4	3-28-63	398 sec.	81 mi.	219 mi.	121 sec.	One inboard engine shut down intentionally after 100 sec. and flight continued successfully.
SATURN IB						
AS-201	2-26-66	1,917 sec.	306 mi.	5,400 mi.	602.9 sec.	Successful suborbital lob shot to position Spacecraft for earth reentry heat shield test.
AS-202	8-25-66	5,582.2 sec.	617 mi.	17,800 mi.	588.5 sec.	Successful suborbital flight to test spacecraft's heat shield, and check launch vehicle.

ORBITAL FLIGHTS

	Date	Perigee	Apogee	Orbital Period	Burn Duration 1st Stg.	Burn Duration 2nd Stg.	Burn Duration 3rd Stg.	Remarks
SATURN I								
SA-5	1-29-64	163 mi.	479 mi.	95 min.	146 sec.	481 sec.	0	First flight with live second stage. 37,900 lbs. into orbit.
SA-6	5-28-64	114 mi.	149 mi.	88 min.	149 sec.	473 sec.	0	Boilerplate Apollo Spacecraft. One inboard engine unexpectedly shut down 26 sec. early but did not impair flight.
SA-7	9-18-64	112 mi.	145 mi.	88 min.	147 sec.	471 sec.	0	Boilerplate Apollo Spacecraft. 39,000 lbs. into orbit. Declared operational three flights early.
SA-9	2-16-65	309 mi.	463 mi.	97 min.	145 sec.	473 sec.	0	First operational flight. Pegasus I placed into orbit.
SA-8	5-25-65	315 mi.	465 mi.	97 min.	148 sec.	473 sec.	0	Pegasus II. First night launch.
SA-10	7-30-65	328 mi.	330 mi.	95 min.	148 sec.	479 sec.	0	Pegasus III.
SATURN IB								
AS-203	7-5-66	*115 mi.	*117 mi.	*88 min.	142 sec.	288 sec.	0	Test liquid hydrogen behavior. Simulation of Saturn V restart conditions.
AS-204	1-22-68	101 mi.	120 mi.	88 min.	453 sec.	0	0	Launched LM on first flight test. Mission successful.

*Orbital parameters given are for initial orbit only; propulsive experiments caused small variations.

both propellants by adjusting the consumption rate of liquid oxygen. Sensors in the vehicle tanks monitor propellant mass throughout the flight and direct an engine-mounted propellant utilization valve to vary the flow of LOX so that it will be depleted simultaneously with the depletion of LH_2. Varying the LOX consumption rate, which also controls engine thrust, caused the vehicle guidance system to compensate by extending the burn time. The variation in LOX consumption rate and burn time both were well within the planned tolerances, and the predicted burnout velocity was achieved. The guidance and control system performed well; both S-IB and S-IVB trajectories and end velocities were normal. No structural problems were found in either of the stages or the instrument unit. The quality of data received at ground stations was good and very few losses occurred in the approximately 1,200 measurements telemetered.

One of the two cameras carried aboard the first stage and ejected following burn was recovered by Air Force crews. The camera had excellent coverage of stage separation and S-IVB ignition. However, neither of the two camera capsules parachutes functioned properly.

The test marked the introduction of a new launch vehicle for the U.S. space program with an unmatched payload capability and with all stages and systems fully functional on the first flight.

SATURN IB AS-203

The second Apollo/Saturn IB (AS-203) added to the impressive record of Saturn vehicles by making the 12th consecutive successful flight in as many launch attempts.

The unmanned vehicle lifted off the pad at launch complex 37 NASA-Kennedy Space Center at 9:53 a.m. (EST) on July 5, 1966.

The AS-203 was topped by a simple aerodynamic shroud (nose cone) instead of an Apollo spacecraft as was carried on the previous Saturn AS-201. The second stage (S-IVB), containing about 10 tons of liquid hydrogen was the "payload." The vehicle instrument unit and nose cone weighed a total of about 58,500 pounds, the heaviest object launched into orbit by the United States.

The primary purpose of the flight was to verify the orbital conditioning characteristics of the second stage propulsion system which uses liquid hydrogen as its fuel. This information was needed for future Saturn V applications in which the S-IVB stage must restart in earth orbit.

Engineers at the NASA-Marshall Space Flight Center, the agency responsible for Saturn development, and Douglas, the second stage contractor, planned the experiment to determine the behavior of liquid hydrogen under weightless conditions. The hydrogen continuous vent system was arranged to provide a very slight amount of thrust as the gaseous hydrogen produced by boiloff escaped. Additional thrust was provided periodically by opening a liquid oxygen tank propulsive vent valve. The information obtained was needed to determine if the thrust and resulting slight acceleration would keep the fuel settled in the bottom of the tank where it would be available for use in restarting the J-2 engine.

Television pictures of the tank interior showed that the fuel settled properly in the bottom of the tank and remained stable during venting, thus verifying the theory. Simulated engine restart conditions were also accomplished successfully.

Launch of AS-203 was delayed one hour and 53 minutes due to trouble in the television system. One of two camera systems was inoperative. Rather than postpone the flight, officials decided to proceed with the remaining camera in operation.

Performance of the first stage was normal. Engine cutoff occurred 141.8 seconds after liftoff; 0.8 seconds sooner than predicted. Stage separation took place at 142.6 seconds. The J-2 engine of the second stage ignited at 144 seconds and cutoff occurred at 432.5 seconds. Cutoff was predicted at 435.3 seconds.

The guidance and control system performed well. The vehicle reached the proper altitude and velocity for insertion into orbit 10 seconds after second stage cutoff.

The orbit planned for AS-203 was circular at an attitude of about 115 statute miles. The first orbit of the vehicle was almost exactly as planned, having a perigee of 115 miles and apogee of 117.6 miles. Orbital period was 88.24 minutes.

Later orbits varied slightly, as expected, because the slight thrust provided by the continuous venting of hydrogen continued to increase the vehicle's velocity, causing the stage to drift slowly into higher orbital paths.

The second stage was broken up near the beginning of the fifth orbit during a hydrogen tank pressure rise rate and bulkhead test. The last telemetry received from the vehicle at the beginning of the fifth orbit indicated the pressure inside the fuel tank was 39.4 psi and the oxygen tank 5 psi. This created a pressure differential across the common bulkhead of 34.4 psi. Shortly after this point a structural failure occurred at the bulkhead, disintegrating the stage. The structural failure of the bulkhead was anticipated, but the time when failure might occur was uncertain.

The failure verified results of a similar test performed on a test model some months earlier at

Douglas in which the common bulkhead failed at very near the same pressure differential. Payload breakup had no effect on the AS-203 mission because all other planned experiments had been completed.

Pressure readings proved that the stage structure would withstand pressure differentials between the two tanks more than three times greater than those experienced under normal operating conditions, further verifying stage design. Another bulkhead pressure test is planned during the next Apollo/Saturn mission (AS-202) after the second stage and spacecraft are separated.

The quality of data received at the ground stations was good and the television pictures were excellent. About 1,500 measurements were telemetered back to earth from the complete vehicle. One of two ejectable motion picture cameras carried in the interstage section was recovered. The color pictures taken by the recovered camera were excellent.

All aspects of the flight including general tests of the vehicle's propulsion and guidance systems and observation of the instrument unit's operation in orbit were carried out satisfactorily.

SATURN IB AS-202

The third unmanned Apollo/Uprated Saturn I (AS-202), the 13th vehicle of the Saturn Program, was launched successfully into space at 12:15:32 p.m. (EST) on August 25, 1966. Liftoff of the vehicle was from launch complex 34 at the NASA-Kennedy Space Center.

Primary purpose of the suborbital flight was to test the Apollo spacecraft's heat shield. Also, the flight provided another check of the launch vehicle.

About 93 minutes after launch, the uprated Saturn had hurled its payload three-fourths of the way around the earth.

The uprated Saturn pushed the spacecraft into space with its 1.6 million-pound thrust first stage, and 200,000-pound thrust second stage before spacecraft separation. Then the Apollo's 21,500-pound thrust service engine carried the spacecraft to an altitude of more than 700 miles.

The Apollo command module made a "skipping" re-entry into the atmosphere, somewhat like a roller coaster ride, subjecting the heat shield to extended high heat loads. The previous re-entry test had been at a sharper angle reducing time of re-entry.

Splashdown of the command module occurred in the Pacific Ocean near the vicinity of Wake Island. The recovery point was about 17,800 miles from the launch site at Kennedy Space Center.

The successful flight followed a three-day delay which allowed engineers to complete minor reworking and additional testing of components in the spacecraft stabilization, guidance, and control systems.

The first stage S-IB performed satisfactorily. Shutdown of the first stage engines occurred at 143.5 seconds, or 1.1 seconds earlier than nominal.

First stage S-IB and second stage S-IVB separation occurred at 144.2 seconds, followed by ignition of the S-IVB stage 1.4 seconds later. Active guidance was initiated successfully 28.2 seconds after separation. All ullage rockets functioned as expected and were jettisoned successfully.

Second stage cutoff occurred at 588.5 seconds, or 13.7 seconds earlier than predicted. Separation of the spacecraft occurred 10.2 seconds after second stage cutoff, or 13 seconds earlier than predicted.

Overall performance of the second stage propulsion system was satisfactory. The J-2 engine was flown at a mixture ratio of approximately 5.5:1 for the first 350 seconds of burn, after which the mixture ratio was changed to approximately 4.7:1. Late mixture ratio cutback contributed to the higher average stage performance. The vehicle liquid hydrogen recirculation valve failed to close as scheduled just prior to J-2 engine start, but the valve failure had no effect on the mission.

The guidance system performed adequately, and the control system deviations were about as expected. Acoustic levels and vibration levels were within expected tolerances and no structural problems appeared in the first stage, the second stage, or the instrument unit. The launch vehicle's electrical systems performed as expected and within appropriate limits.

The emergency detection system (EDS) was flown "closed loop" on this flight. The overall operation of the EDS was successful. However, an intermittent electrical short circuit beginning at 93.6 seconds occurred in the "Q-ball," which is an EDS sensor. Finally, a hard short occurred at 114.9 seconds, in a regulator circuit. Other portions of the launch vehicle's EDS performed properly.

Exhaust gas from the spacecraft propulsion system (SPS), which ignited 11 seconds after separation, caused the second stage to oscillate. Also, two of the spacecraft-lunar, module-adapter panels either came off or folded back due to the SPS exhaust forces on them. However, the second stage's auxiliary propulsion system (APS) brought the disturbances under control.

Mission evaluation proved flight AS-202 was successful, and met all expected requirements.

SATURN IB AS-204

The fourth unmanned Apollo/Saturn IB (AS-204), the 14th vehicle of the Saturn I program, was launched successfully at 5:48 p.m. (EST) on January 22, 1968, from launch complex 37 at the NASA Kennedy Space Center.

Primary purposes of the earth orbital flight were to verify operation of the propulsion systems of the descent and ascent stages of the Apollo lunar module (LM), including restart, to evaluate LM staging and LM structures and to further evaluate launch vehicle performance. A propellant dumping experiment was conducted with the launch vehicle second stage after separation from the spacecraft.

The Apollo command and service modules were not flown on AS-204. The LM was enclosed during early flight in a spacecraft/lunar module adapter (SLA) topped by an aerodynamic nosecone.

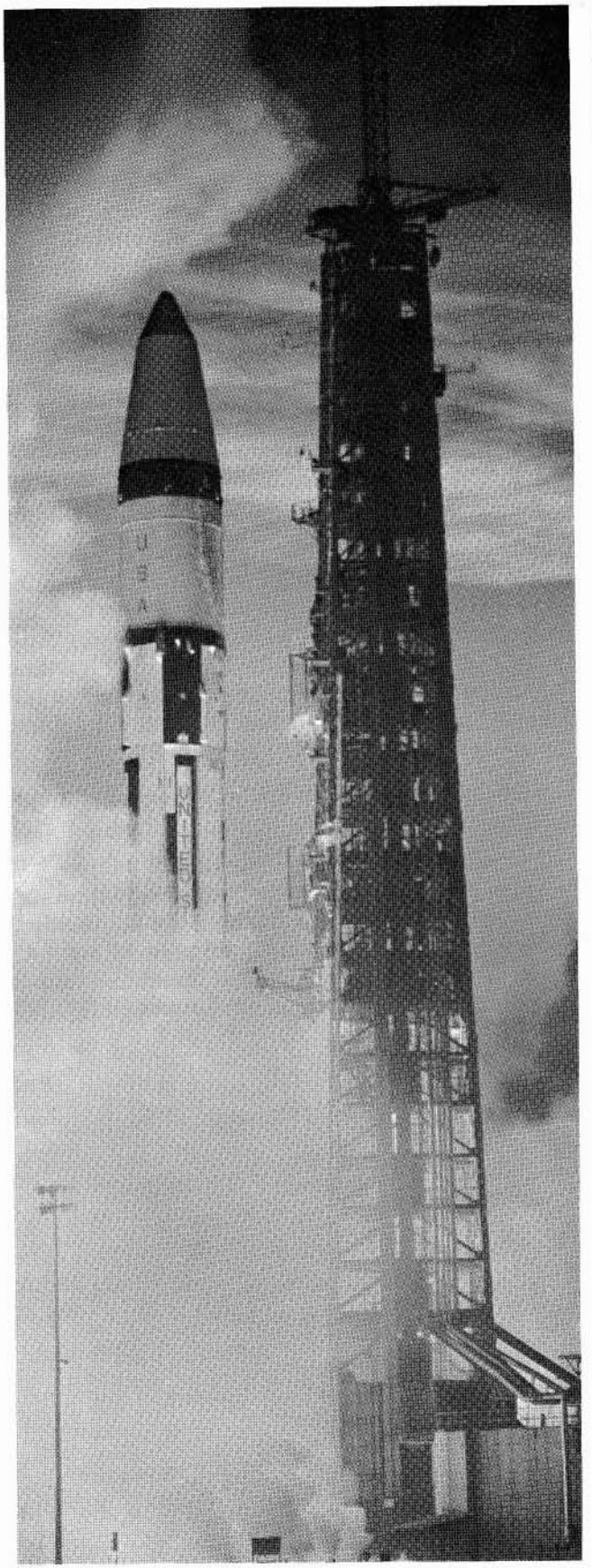
The Saturn IB placed the payload into orbit at an altitude of 101.4 statute miles, 1,142.7 miles downrange from Cape Kennedy.

The launch vehicle reached Mach 1 at 59.8 seconds after liftoff, 0.4 second early, at an altitude of 4.66 statute miles. Maximum dynamic pressure was reached at 71.5 seconds, 2.7 seconds earlier than expected, at an altitude of 7.13 statute miles, about 0.54 miles lower than nominal.

The first and second stages separated at 2 minutes, 23.5 seconds after launch, just 0.1 second early. Second stage cut-off came at 9 minutes, 53 seconds. Trajectories of both stages were within expected tolerances, and orbit insertion conditions were met. Orbital velocity was 17,515.3 miles per hour.

No anomalies or system deficiencies were noted in the first stage. Fuel pressurization system operation was normal, and pressure was adequate to maintain structural rigidity. The LOX pressurization system performed satisfactorily, gaseous oxygen flow control valve movements were as anticipated, and control pressure system operation was adequate. Hydraulic system performance was also adequate. Retrorocket ignition of all four units was normal, and burn times were nominal.

Performance of second stage propulsion and associated systems was nominal. The new liquid hydrogen recirculation chilldown control valve performed normally. All J-2 engine start and run requirements were met. Prepressurization and pressurization in the propellant systems were normal. At J-2 start the LOX ullage pressure was within the predicted band. The slightly shorter



Saturn IB AS-204

S-IVB stage burn time was partially due to a slightly higher-than-predicted average thrust during the mainstage burn.

The vehicle's guidance system performed satisfactorily and no gimbal angle failures occurred. All switch selector functions available in telemetry were issued within the 50 millisecond tolerance envelope. The S-IVB propellant mixture ratio shift was detected 27.3 seconds later than the predicted time but well within the expected tolerance of plus or minus 45 seconds. The control system performed as expected. The auxiliary propulsion system functioned well. APS propellants remaining in the system about three hours after orbit insertion, and after completion of the propellant dump experiment, were greater than expected.

Maximum rates in pitch were between plus 1.1 and minus 1 degree at liftoff, yaw rates were negligible, and the roll rate was plus 2 degrees.

Base and aerodynamic heating environments were within expected limits. Vehicle environmental pressures were within expected tolerances and the instrument unit thermal conditioning system pressures and flowrate were within proper ranges. Performance of the measuring systems on all stages appeared to be excellent. No definite inflight measurement failures were identified.

The emergency detection system (EDS) was flown in open-loop mode.

The nosecone carried atop the launch vehicle was ejected, and the SLA panels were deployed, exposing the LM. The first burn of the LM descent propulsion system came at 59 minutes, 40 seconds, following separation of the LM from the launch vehicle. The burn was cut off prematurely after 4.3 seconds, due to overly conservative programming of the LM guidance computer. This prompted a switch to an alternate mission plan for the balance of the LM experiments, including two more DPS burns and two burns of the ascent propulsion system.

The S-IVB liquid oxygen dump experiment began at 2 hours, 26 minutes, and 14 seconds into the flight, and ended 2 minutes later. At 2 hours, 28 minutes and 24 seconds, the liquid hydrogen dump began, ending after 3 minutes, 4 seconds. In the experiments, propellants remaining in the stage after cutoff were exhausted through the J-2 engine in a test to determine the vehicle disturbances imparted by the flow of propellants.

The thrust vector control system adequately controlled disturbances caused by the LOX dump. Maximum attitude errors recorded were about 5.5 degrees in the pitch and yaw planes. The auxiliary propulsion system was used to control attitude changes caused by the liquid hydrogen dump. Maximum attitude errors were about one degree in the pitch and yaw planes. The dumping of residual gases in the cold helium sphere and the ambient helium bottle, also part of the experiment, was initiated on schedule.

The last good signal contact from the second stage and the instrument unit occurred at Tananarive, Australia, at 10 hours, 1 minute into the flight. The stage apparently reentered the atmosphere at about 9:20 a.m. EST on January 23 in the southwest Pacific.

Mission evaluation showed that the AS-204 flight was successful and that all mission requirements were met.

Appendix A—Glossary of Terms

The following list defines acronyms, abbreviations, nomenclature, and other terminology used in the Saturn IB News Reference.

TERM	DESCRIPTION
A	Ampere
ABMA	Army Ballistic Missile Agency
Acceptance Test	A test, or series of tests, conducted in accordance with Government specification on each production engine or stage to demonstrate that the performance of that engine or stage and its components throughout all tests is within the limits specified in the model specification.
Actuators	Devices which transform an electrical signal into a mechanical motion using hydraulic or pneumatic power.
AES	Apollo Extension System
AGC	Automatic Gain Control
Apollo	Designation for America's manned lunar landing project; also, designates the spacecraft used for manned lunar landing.
APS	Auxiliary Propulsion System
Area Ratio (ϵ)	See expansion area ratio
Arm To Arm	As applied to ordnance, changing from a safe condition to a state of readiness for initiation or ignition.
ARPA	Advanced Research Projects Agency
ASI	Augmented Spark Igniter
Ballistic Missile	A guided missile, the major portion of whose flight path to its target is a ballistic trajectory.
Ballistic Trajectory	The flight path of a vehicle, projectile or bomb which is determined by gravitational and drag forces alone. Specifically, propulsion thrust forces and aerodynamic lifting forces are either absent, negligible or used for only minor corrections of the flight path.
Blow-Out-Disc	A mechanism, consisting of a thin metal diaphragm, used as a safety device to relieve excessive gas pressure.
Boilerplate	A full-size mockup that has all of the mechanical characteristics of the true item but none of the functional features.
Booster	A part of the propulsion system of an air vehicle, usually a rocket propulsion system, used to accelerate an air vehicle from its launching velocity to flight velocity. The booster may or may not separate from the air vehicle when its action is completed.
Bootstrap	A self-generating or self-sustaining process. During mainstage operation of a rocket engine, the gas generator is fed by the main propellants pumped by the turbopump. The turbopump in turn is driven by hot gases from the gas generator system. Such a system must be started by a starting system which supplies outside power or propellants. When rocket-engine operation is no longer dependent on outside power or propellants, it is said to be in "bootstrap" operation.
Bridge Wire	Wire in ordnance item which heats to a high temperature and burns, thus igniting a charge. It "bridges" the gap between the electrical conducting wire and the charge.
Burst Diaphragm	See blow-out disc
Cavitation	The formation of a partial vacuum within a pump caused by irregular distribution of fuel within the pump cavity.
CCSD	Chrysler Corporation Space Division, prime contractor for the Saturn S-IB stage

TERM	DESCRIPTION
CDF	Confined Detonator Fuse
Chamber Pressure (P_c)	The pressure (psia) in the combustion chamber of a rocket engine.
Combustion	A chemical process characterized by the evolution of heat. Commonly, the chemical reaction of fuel and oxidizer, but by extension includes the decomposition of monopropellants.
Combustion Chamber	An enclosed volume within which a controlled combustion takes place.
Combustion Chamber Length	The length (in.) of the combustion chamber from the injector face to the section through the nozzle throat.
Component	A term loosely given to a part, subassembly, assembly, or system, when it is considered as part of a larger assembly or system.
Component Test	A test of any type performed upon a part, assembly, or system.
Cutoff	The shutting-off of a working fluid or a combustion process. Specifically, in a liquid-propellant rocket engine, the time when either one or both of the main propellant valves is completely shut off. The cutoff is normally followed by tail-off. (Ref: MIL 5150A (ASG))
DDAS	Digital Data Acquisition System
DE/Q	Design Evaluation and Qualification
Development	The application of known scientific facts, techniques, materials, and physical laws to the creation of new or improved material or methods.
Development Test	A test made by the developing agency to verify the operation or performance of a system or component design, or to produce data which will permit improving the design of the item under test.
Doppler Shift (or Effect)	A phenomenon by which the radio signal received from a vehicle appears higher in frequency as the vehicle approaches the radio receiver and lower in frequency as the vehicle recedes from the receiver.
Duration, Effective	Of a rocket engine, the time (sec) of mainstage operation.
Duration, Shutdown	The interval between the time of the shutdown signal to the time at which the thrust of the rocket engine is negligible.
Duration, Total	Of a rocket engine, the total firing time (sec) from the beginning of the thrust rise to the end of the thrust tail-off.
EBW	Exploding Bridgewire
EDS	Emergency Detection System
Effective Propellant	The total impulse (lbf sec) divided by the effective specific impulse (lbf sec/lbm). In effect it is the total propellant less the propellant lost in starting and shutdown, and includes propellant trapped in tanks, pumps, lines, valves, cooling jackets, etc.
Engine	An independent, self-contained unit for supplying thrust or propulsive power to an air or space vehicle. It ordinarily does not include the following systems: main propellant supply system, pneumatic power supply system, or electrical power supply system. Accordingly, any assembly of parts, assemblies, or systems is termed an engine if it can produce thrust or propulsive power given only the inputs of the air or space vehicles main propellant supply system, pneumatic power supply system, and electrical power supply system.
Environment	The aggregate of the conditions and forces that influence or modify an article and its performance throughout its life history. Environment is contingent upon the operation for which

TERM	DESCRIPTION
Environment (continued)	the article is designed and in turn determines many details of the design. It includes both natural and induced phenomena. See environment, natural; environment, induced. (Ref: ARDCM 80-1)
ETR	Eastern Test Range
Exhaust Nozzle	That portion of the nozzle of a thrust chamber which lies on the downstream side at the nozzle throat.
Expansion Area Ratio (ϵ_θ)	The ratio of the nozzle exit section internal area to the nozzle throat area.
FM/DDAS	Frequency Modulated/Digital Data Acquisition System
FM/FM	Frequency Modulated/Frequency Modulation
GH_2	Gaseous Hydrogen
Gimbal	A mechanical frame containing two mutually perpendicular intersecting axes of rotation (bearings and/or shafts).
GN_2	Gaseous Nitrogen
GOX	Gaseous Oxygen
gpm	Gallons per minute
GSE	Ground Support Equipment
H-1	Saturn IB stage engine
Heat Exchanger	A device which provides means for the transfer of heat from one fluid or gas to another.
Hypergolic	Refers to those bi-propellant combinations which ignite spontaneously upon contact or mixing.
IBM	International Business Machines
Igniter	A device used to initiate burning of a propellant mixture.
Ignition Stage	The portion of the starting sequence wherein combustion is initiated and stabilized at a low level in the thrust chamber. In the H-1 and J-2 engines, flow of the oxidizer from the main oxidizer feed system into the combustion chamber is established under gravity head alone and fuel is supplied from the start system. The mixture is ignited by the igniter.
Inducer	A portion of a centrifugal pump which increases the pressure and whirl of the fluid before it reaches the main impeller.
Inertial Guidance	A self-contained, automatic system which uses gyroscopic devices to interpret information on speed, attitude, etc., and adjusts the vehicle to a predetermined flight path.
Injector	A device designed to introduce propellant(s) into the combustion chamber.
Interface	Common boundary between two or more components, systems, stages, etc.
IRBM	Intermediate Range Ballistic Missile
IRIG	Inter-Range Instrumentation Group
KSC	Kennedy Space Center
LC-34	Launch Complex 34, Cape Kennedy
LEM	Lunar Excursion Module
LH_2	Liquid Hydrogen
LN_2	Liquid Nitrogen
LOR	Lunar Orbital Rendezvous
LOX	Liquid Oxygen
LVDA	Launch Vehicle Data Adapter
LVDC	Launch Vehicle Digital Computer
MAF	Michoud Assembly Facility, New Orleans

TERM	DESCRIPTION
Mainstage	The operation of a liquid-propellant rocket engine at or above 90 per cent of rated thrust.
MDF	Mild Detonating Fuse
Michoud	NASA manufacturing complex in New Orleans, jointly occupied by NASA and several contractors currently working on project Apollo.
MILA	Merritt Island Launch Area
Mixture Ratio (r)	The ratio of the oxidizer consumption rate (lbm/sec) to the fuel consumption rate (lbm/sec).
Monocoque	A type of construction where the structural loads are transmitted through the surface covering or skin rather than a frame.
MSFC	Indicates the Marshall Space Flight Center, whose headquarters are in Huntsville, Alabama, or any of its branch facilities.
Multiplexer	A device which collects data from many sources and arranges it for simultaneous transmission over a single network.
NASA	National Aeronautics and Space Administration
Net Positive Suction Head (NPSH)	A positive supply of fuel with sufficient pressure to supply the propellant pumps.
ODOP	Offset Doppler Tracking System
Orifice	A fixed restriction in a fluid passage which establishes the rate of fluid flow. Often an orifice can be changed as means for calibration of some fluid-flow parameter. It usually consists of a thin circular diaphragm with an accurately machined concentric hole.
Palaemon	Barge for shipping Saturn IB first stages and other large vehicle components over water routes.
PAM	Pulse Amplitude Modulation
PCM/FM	Pulse Code Modulated/Frequency Modulation
Pitch	Displacement of the vehicle from its lateral axis.
Pressure, Stagnation (p_t)	The theoretical pressure that would be realized if a moving fluid stream could be brought to rest with no loss in energy, either internal or external.
Prevalves	Leakproof valves which, when open, permit the propellant to reach the turbopumps prior to engine ignition.
Primacord	An explosive charge, shaped like a rope, part of the range safety propellant dispersion system.
Promise	Barge transporter for Saturn IB first stage and other large vehicle components.
Propellant	A liquid or solid substance or substances which either separately or mixed can be changed into a large volume of hot gases at a rate which is suitable for propelling projectiles or air vehicles. Propellants may be used either in guns or rocket propulsion systems.
Propellant Feed System	The portion of a liquid-propellant rocket engine which moves the main propellants under calibrated or controlled conditions from the propellant supply system to the thrust chamber assembly. It includes the piping, sensors, main propellant valves, associated actuators, and turbopump assembly (if any).
Propulsion System	The portion of an air or space vehicle whose function is to provide thrust. This thrust is normally used to propel the vehicle through all phases of powered flight, but may be used for attitude control purposes. These control thrusts may produce no propelling force. In general it includes the engines and all necessary associated systems to ensure satisfactory operation of the engine either on the ground or in flight.
psi	Pounds per square inch

TERM	DESCRIPTION
psia	Pounds per square inch absolute
psig	Pounds per square inch gage
PU	Propellant Utilization
R&D	Research and Development
REAP	Rocket Engine Advancement Program
Regenerative Cooling	A system for keeping liquid rocket engines cool in which one of the liquid propellants is circulated through the engine thrust chamber walls to protect the walls under high combustion temperatures.
RF	Radio Frequency
Rocket	A rocket engine, a rocket motor, a rocket propulsion system, or a complete vehicle or projectile which uses the principles of rocket propulsion.
Roll	The rolling of a vehicle about its axis as the vehicle continues on its flight path.
RP-1	Kerosene like fuel
rpm	Revolutions per minute
S-IB	Chrysler-built first stage of Saturn IB vehicle.
S-IVB	Douglas-built second stage of Saturn IB vehicle.
SARAH	Search and Rescue and Homing
Saturn I	First, large, space vehicle preliminary to the moon flight vehicle.
scfm	Standard cubic feet per minute
SCO	Subcarrier Oscillators
Sequencer	A mechanical or electronic device that initiates a series of events to make the events occur in a particular sequence.
Shutdown	The process by which the thrust of a rocket engine is brought from its mainstage, steady-state value to a negligible value. To be distinguished from cutoff which refers particularly to the operation of the main propellant valves.
Shutdown Sequence	The series of events by which shutdown is accomplished. In liquid propellant rocket engines it typically involves (1) shutting off pump power source and/or throttling main propellant valves to decrease propellant flow to thrust chamber, (2) cutoff of propellant flow, (3) tail-off of residual propellants and gases.
Simulated Flight Test	A test whereby the stage is programmed through a complete flight and flight sequence to check out all circuits and events without having propellant flow or igniting ordnance items.
Specific Impulse (I_s)	The impulse (lbf/sec) delivered by a rocket propulsion system in any time interval divided by the mass (lbm) or propellants consumed during the effective duration (sec).
Spider Beam	A spider-web-like frame which holds the top of the propellant tanks together, transmits thrust to the upper stages, and adapts the S-IB stage to the upper stages.
Squib	A small explosive device whose primary function is to produce heat; usually used to achieve ignition in a larger combustion process.
SS/FM	Single Sideband/Frequency Modulated
Stage	A period of operation under steady-state conditions occurring between operating discontinuities. Specifically, it refers to those portions of powered flight of an air or space vehicle, in which both the thrust and the mass of the vehicle are either essentially constant or are varying slowly and smoothly. A discontinuity of either thrust or mass could signal the end of a stage. Generally the discontinuity is caused by the separation of an engine or propellant tankage, or both.

TERM	DESCRIPTION
Stage (continued)	Also used to characterize such separable portions of an air or space vehicle. A steady-state condition of propellant flow and combustion in a rocket engine. Often used to refer to portions of the start sequences leading to mainstage operation.
Tanks, Propellant	The portion of the propellant supply system which consists of necessary structure and accessories to form a gas-tight container with adequate provisions for filling, emptying, venting, purging or pressurizing under conditions stipulated. In general, the tanks will either be integral or removable. (Ref: ARDCM 80-1)
Test	The operation of an assembly or system under conditions which simulate in some manner or degree the conditions under which that item is expected to operate in service for the purpose of obtaining observations or measurements relating to its manner of operation.
Throat	The portion of a fluid passage where the cross-sectional area is less than the cross-sectional area on either side of that portion.
TMR	Triple Modular Redundancy
Transducer	A device which converts one form of energy into another. For example, a loudspeaker is a transducer which converts electrical energy into acoustical energy.
Turbine	The prime mover (powered by hot gases from the gas generator system) in the turbopump assembly which, through a shaft and/or gear train, drives the main propellant pumps.
ULD	Unit Logic Device
Ullage	The volume in a closed tank or container above the surface of a stored liquid. Also the ratio of this volume to the total volume of the tank. Due to differential expansion rates between the stored liquid and the tank material, the ullage is a function of temperature.
Venturi Tube	A tube with a constricted throat, which by virtue of the reduced flow pressure at the constriction, is used in gages, injectors, gyroscopes, etc.
Yaw	Displacement of a vehicle from its longitudinal axis.

Appendix B – Saturn IB Subcontractors

The following are lists of subcontractors who have played a major role in the development and production of the Saturn IB launch vehicle. It should be recognized that many more subcontractors contributed to the total vehicle and program, however it is not practical to list all in this document.

S-IB STAGE MAJOR SUBCONTRACTORS

SUBCONTRACTOR	LOCATION	PRODUCT
Acoustic Associates, Inc.	Los Angeles, Calif.	Sensors
Adel Products, Div. General Metals Corp.	Huntington, W. Va.	Fasteners
Aeronca	Middleton, Ohio	Sheet Metal Components
Airite, Div. of Sargent Ind.	El Segundo, Calif.	Titanium Vessels
Arrowhead Products, Div. Federal-Mogul Bower Bearings, Inc.	Los Alamitos, Calif.	Metal Bellows Ducting
Automation Industries, Inc.	El Segundo, Calif.	Profile Machined Parts
Avnet Electronics Corp.	Franklin Park, Ill.	Electronic Equipment
Benson Mfg. Co.	Kansas City, Mo.	Sheet Metal Components
Bourns Inc., Instrument Div.	Riverside, Calif.	Electronic Equipment
Cadillac Gage Co.	Detroit, Mich.	H-1 Engine Hydraulic Systems
Cosmodyne Corp.	Hawthorne, Calif.	Electronic Equipment
Dupont de Nemours E & I Co.	Wilmington, Del.	Chemical Products
Electronic Associates, Inc.	Long Branch, N.J.	Electronic Equipment
Electronic Div., DIT-MCO, Inc.	Kansas City, Mo.	Electronic Equipment
Electronic Div., Ralph Parson Co.	Pasadena, Calif.	Electronic Equipment
Emerson Electric Mfg. Co.	St. Louis, Mo.	Thermal Insulation
Engineered Magnetics Div., Gulton Ind.	Hawthorne, Calif.	Electronic Equipment
Fenwal, Inc.	Ashland, Mass.	Probe Assemblies
General Electric Corp.	Houston, Texas	Electric Equipment
Goodyear Aerospace Corp.	Akron, Ohio	Seal Plates and Panels
Gulton Ind., Inc.	Metuchen, N.J.	Accelerometers
Hayes International Corp.	Birmingham, Ala.	Structures
Hewlett-Packard Co.	Palo Alto, Calif.	Test Equipment
Hi-Shear Corp.	Torrance, Calif.	Fasteners
Honeycomb Co. of America	Bridgeport, Conn.	Honeycomb Paneling
Huck Mfg. Co.	Detroit, Mich.	Fasteners
International Data Systems, Inc.	Dallas, Texas	Electronic Equipment
Ladish Co.	Cudahey, Wisc.	Forgings
Lockheed-Georgia Co.	Marietta, Ga.	Structures
LTV Aerospace Corp.	Dallas, Texas	Propellant Containers
Marotta Valve Corp.	Boonton, N.J.	Valves
MB Electronic Div., Textron Electronics	New Haven, Conn.	Test Equipment
Military Products Div., Clary Corp.	San Gabriel, Calif.	Valves
Moog Servo Controls, Inc.	East Aurora, N.Y.	Hydraulic Actuators
Motorola Inc., Military Electronics Div.	Scottsdale, Ariz.	Transponders
NTW Missile Engineers, Inc.	Los Angeles, Calif.	Precision Machinery
Parker Aircraft Co.	Los Angeles, Calif.	Valves and Fittings
Pioneer Central Div., Bendix Corp.	Davenport, Iowa	Sensing Probes

S-IB STAGE MAJOR SUBCONTRACTORS (Continued)

SUBCONTRACTOR	LOCATION	PRODUCT
Scientific Data Systems, Inc.	Santa Monica, Calif.	Computer Systems
Servomechanisms Inc., Div. of Teledyne Systems	El Segundo, Calif.	Electronic Equipment
Servonic Instruments, Inc.	Costa Mesa, Calif.	Electronic Equipment
Solar Div., International Harvester Co.	San Diego, Calif.	Tubing
Southwestern Ind., Inc	Los Angeles, Calif.	Switches
Space Craft, Inc.	Huntsville, Ala.	Electronic Equipment
Statham Instrument Inc.	Los Angeles, Calif.	Electronic Equipment
Stratas Div., Fairchild Hiller Corp.	Manhattan Beach, Calif.	Valves
Tektronic, Inc.	Houston, Texas	Electronic Instrumentation
Univac, Div. of Sperry Rand	New Orleans, La.	Computer Spare Parts
Wyman Gordon Co.	Worcester, Mass.	Forgings

INSTRUMENT UNIT SUBCONTRACTORS

AVCO Corporation	Nashville, Tenn.	Cable tray, thermal conditioning panels
Bendix Corporation, Eclipse-Pioneer Division	Teterboro, N.J.	Inertial platform and sensing equipment
Brown Engineering Company	Huntsville, Ala.	PCM ground checkout station
Consolidated Systems	Pomona, Calif.	Command guidance decoder
Electronic Communications, Inc.	St. Petersburg, Fla.	Flight control computer
Electro Development Corp.	Seattle, Wash.	DC amplifiers
Hamilton Standard	Windsor Locks, Conn.	Sublimator, water and water/methanol accumulators
Hayes International	Huntsville, Ala.	Network and adapter cables
International Business Machines Corporation, Federal Systems Division	Owego, N.Y.	Launch vehicle digital computer/launch vehicle data adapter
Martin Company	Orlando, Fla.	Control signal processor
Milgo	Miami, Fla.	Analog computers for mission simulation lab.
Motorola Corporation	Phoenix, Ariz.	C-band radar transponder
North American Rockwell	Tulsa, Okla.	Structure segments
Northrop Corporation, Nortronics Division	Norwood, Mass.	Rate gyro package
Statham Instrument	Los Angeles, Calif.	Control accelerometers
Thompson-Ramo-Wooldridge	Cleveland, Ohio	Coolant pump

H-I ENGINE AND J-2 ENGINE MAJOR SUBCONTRACTORS

A & M Castings, Inc.	Southgate, Calif.	Volute
Ace Industries	Santa Fe Springs, Calif.	Rotors & stators, nozzles, bearing support, inducers
Alcoa	Vernon, Calif.	Dome forging
Arcee Foundry	Norwalk, Calif.	Castings
The Buehler Corporation Indiana Gear Works Division	Indianapolis, Ind.	Gears, shafts
Cameron Iron Works	Houston, Texas	Injector forging, forgings
Consolidated Electrodynamics Corp.	Pasadena, Calif.	Pressure transducers
D-K Manufacturing Company, Aerospace Division	Batavia, Ill.	Bellows
Ex-Cell-O Corporation	Detroit, Mich.	Gimbals
Herlo Engineering Company	Hawthorne, Calif.	Valve components, valves, machine parts
Hills-McCanna Corporation	Chicago, Ill.	Gear case castings
Hupp Aviation Div. of Hupp Corporation	Chicago, Ill.	Turbopump gears
La Mesa Tool & Manufacturing, Inc.	El Cajon, Calif.	Machine ring
Langley Corporation	San Diego, Calif.	Gimbals
LeFiell Manufacturing Company	Santa Fe Springs, Calif.	Tubes

H-I ENGINE AND J-2 ENGINE MAJOR SUBCONTRACTORS (Continued)

SUBCONTRACTOR	LOCATION	PRODUCT
L. A. Gauge Company, Inc.	Sun Valley, California	Valve details
Motorola, Inc. Semiconductor Products Div.	Phoenix, Arizona	Electronic components
Precision Sheet Metal, Inc.	Los Angeles, Calif.	Jackets, tubes
Quadrant Engineering Corp.	Gardena, Calif.	Dome machining
Reisner Metals, Inc.	Southgate, Calif.	Rotor forging, forgings
Sealol, Incorporated	Providence, Rhode Island	Seals
Solar Division of International Harvester Company	San Diego, Calif.	Valve components, bi-metal braze, heat exchanger
Southwestern Industries	Los Angeles, Calif.	Pressure switches
Standard Tool & Die	Los Angeles, Calif.	Machine exit manifold ring
Statham Instruments	Los Angeles, Calif.	Pressure transducers
Superior Tube Company	Norristown, Pa.	Tubing
H. I. Thompson Fiber Glass Co.	Gardena, Calif.	Insulation start tank
Trent Tube Company Division of Crucible Steel Co. Of America	East Troy, Wisconsin	Tubing
Turbo Cast, Incorporated	Los Angeles, Calif.	Blades
Union Carbide Corporation, Stellite Division	Kokomo, Indiana	Turbine blades Rene sheet Hastelloy sheet Inco 718
V & W Aircraft Castings	Bell Gardens, Calif.	Castings
Viking Forge & Steel Company	Albany, California	Rolled forged rings
Weldfab, Incorporated	Gardena, California	LOX joint
Western Way, Incorporated	Van Nuys, California	Heat exchanger, hood aspirator
Winsco Instruments & Controls	Santa Monica, Calif.	Temperature transducer
Wyman-Gordon Company	Worcester, Mass.	Injectors, manifolds, throat rings

S-IVB STAGE MAJOR SUBCONTRACTORS

Accessory Products Company, Div. of Textron, Inc.	Whittier, California	Valves/Heaters
Aeroquip Corp., Marman Division	Los Angeles, Calif.	Clamps
Airdrome Parts Company	Inglewood, Calif.	Fittings
Airesearch Mfg. of Arizona	Phoenix, Arizona	Valves
Airtex Dynamics, Inc.	Compton, Calif.	Tank Assem.
Amco Engineering Co.	Chicago, Illinois	Cabinets
American Electronics, Inc.	Fullerton, Calif.	Batteries
Amp, Inc.	Harrisburg, Pa.	Electrical Panels
Ampex Corp.	Los Angeles, Calif.	Tape Recorders
Amphenol Borg Electronics Corp.	Chicago, Illinois	Connectors
Anaconda Metal Hose Div.	Waterbury, Conn.	Metal Hose
Astrodata, Inc.	Anaheim, Calif.	Telemetry
Avnet Corp.	Westbury, L.I., N.Y.	Electrical Connectors
Barry Controls	Watertown, Mass.	Electronic Controls
Bertea Products	Pasadena, Calif.	Transmitter/Fabricated Assem.
Brown Engineering Co., Inc.	Huntsville, Alabama	Telemetry Equipment
Calmec Mfg. Company	Los Angeles, Calif.	Valves
Capital Westward, Inc.	Paramount, Calif.	Filters
Christie Electric Corp.	Los Angeles, Calif.	Meters
Consolidated Electrodynamics	New York. N.Y.	Electronic Equipment

S-IVB STAGE MAJOR SUBCONTRACTORS (Continued)

SUBCONTRACTOR	LOCATION	PRODUCT
Data Sensors	Gardena, Calif.	Transducers
Deutsch Company	Los Angeles, Calif.	Fittings
Dynatronics, Inc.	Orlando, Florida	Telemetry Equipment
Eagle Pitcher Co.	Joplin, Mo.	Batteries
Electra Scientific Corp.	Fullerton, Calif.	Transducers
Electrada Corp.	Culver City, Calif.	Electrical Components
Fairchild Camera Inst. Corp.	Plainview, L.I., N.Y.	Cameras, Oscilloscopes
Fairchild Controls Div.	Montebello, Calif.	Valves
Fairchild Hiller Corp.	Bayshore, L.I., N.Y.	Valves
Fairchild Semiconductor	Hollywood, Calif.	Semiconductors
Fairchild Stratos	Bayshore, L.I., N.Y.	Valves
Flexible Metal Hose Mfg. Co.	Costa Mesa, Calif. Northridge, Calif.	Metal Hose
Flomatics, Inc.	Natoma, Calif.	Valves
Frebank Company	Glendale, Calif.	Switches
Control Data Corp.	Minneapolis, Minn.	Computers
General Electric Co.	Waterford, N.Y.	Electrical Components
Giannini Controls Corp.	Duarte/Pasadena, Calif.	Transducers
Grove Valve Regulator Co.	El Segundo/Oakland, Calif.	Valves
Hadley, B. H. Co.	Pomona, Calif.	Relays
Hewlett Packard Co.	Pasadena, Calif.	Oscilloscopes/Recorder
Hexcel Products, Inc.	Berkeley, Calif.	Honeycomb Panels
Honeywell, Inc.	Minneapolis, Minn.	Temperature Controls, Gyros, Sensors
ITT Wire and Cable	Clinton, Mass.	Wire and Cable
K-Tronics	Los Angeles, Calif.	Semiconductors
Kaiser Aluminum Chemical Sales, Inc.	Spokane, Wash. Halethorpe, Md.	Raw Material, Cable
Kinetics Corp.	Solana Beach, Calif	Electronic Equipment
Ladewig Valve	Los Angeles, Calif.	Valves
Lanagan, W. M. Co., Inc.	Costa Mesa, Calif.	Valves
Leonard, Wallace O., Inc.	Pasadena, Calif.	Valves
Linde Co., Div. Union Carbide	El Monte, Calif.	Batteries
Litton Industries of Calif.	Beverly Hills, Calif.	Transducers
Magnesium Alloy Prods. Co.	Compton, Calif.	Castings
Magnetika, Inc.	Venice, Calif.	Batteries
Marotta Valve Corp.	Santa Ana, Calif.	Valves
Marshall, G. S. Company	San Marino, Calif.	Electronic Hardware
Mason Electric Div. Ansul	Los Angeles, Calif.	Switches
Master Specialties Co.	Los Angeles, Calif.	Meters
Menasco Mfg. Company	Burbank, Calif.	Fabricated Assem.
Military Products Div. Clary	San Gabriel, Calif.	Elec. Fittings
Moog Servo Controls, Inc.	Aurora, N.Y.	Valves
Motorola Semiconductor Products	Hollywood, Calif.	Semiconductors
Pacific Valve, Inc.	Long Beach, Calif.	Valves
Parker Aircraft Company	Los Angeles, Calif.	Fittings/valves
Pesco Products Div. Borg Warner	Bedford, Ohio	Pumps

S-IVB STAGE MAJOR SUBCONTRACTORS (Continued)

SUBCONTRACTOR	LOCATION	PRODUCT
Philco Corp.	Philadelphia, Pa.	Radio Equipment
Planautics Corp.	Solana Beach, Calif.	Switches
Potter Brumfield	Princeton, Ind.	Switches, Relays
Purolator Products, Inc. Western Div.	Van Nuys, Calif.	Filters
Reynolds Metals Co.	Birmingham, Ala.	Raw Material (Alum.)
Rosemount Eng. Co.	Minneapolis, Minn.	Temperature Controls
Sandorn Co.	Waltham, Mass.	Recorders
Scintilla Div. Bendix	Sidney, N.Y.	Electrical Connectors
Sealol Corp.	Providence, R.I.	Valves
Servonic Instruments, Inc.	Costa Mesa, Calif.	Transducers
Signet Scientific	Burbank, Calif.	Ovens
Snap Tite	Union City, Pa.	Connectors
Sperry Gyro Co. Div. Sperry Rand	Great Neck, L.I., N.Y.	Instruments
Stainless Steel Products	Burbank, Calif.	Flexible Ducts
Statham Instruments, Inc.	Los Angeles, Calif.	Transducers
Trans-Sonics, Inc.	Lexington, Mass.	Temperature Controls
Thompson-Ramo-Wooldridge	Cleveland, Ohio	Attitude Control Engines
Technology Instruments Corp.	Newbury Park, Calif.	Potentiometers
Telemetrics, Inc., Subsidiary of Arnoux Corp.	Santa Ana, Calif.	Ground Support Electronics
Texas Instrument, Inc.	Dallas, Texas	Resistors/Transistors
U.S. Steel Corp. U.S. Steel Supply Div.	Seattle, Wash.	Raw Material
Vacco Valve Company	El Monte, Calif.	Valves
Vickers	Detroit, Mich.	Pumps
Vinson Mfg. Co., Inc.	Van Nuys, Calif.	Valves
W & S Industries	El Monte, Calif.	Connectors
Winsco Instruments Control	Santa Monica, Calif.	Temp. Control Units
Wyman Gordon Company	Worcester, Mass.	Forgings

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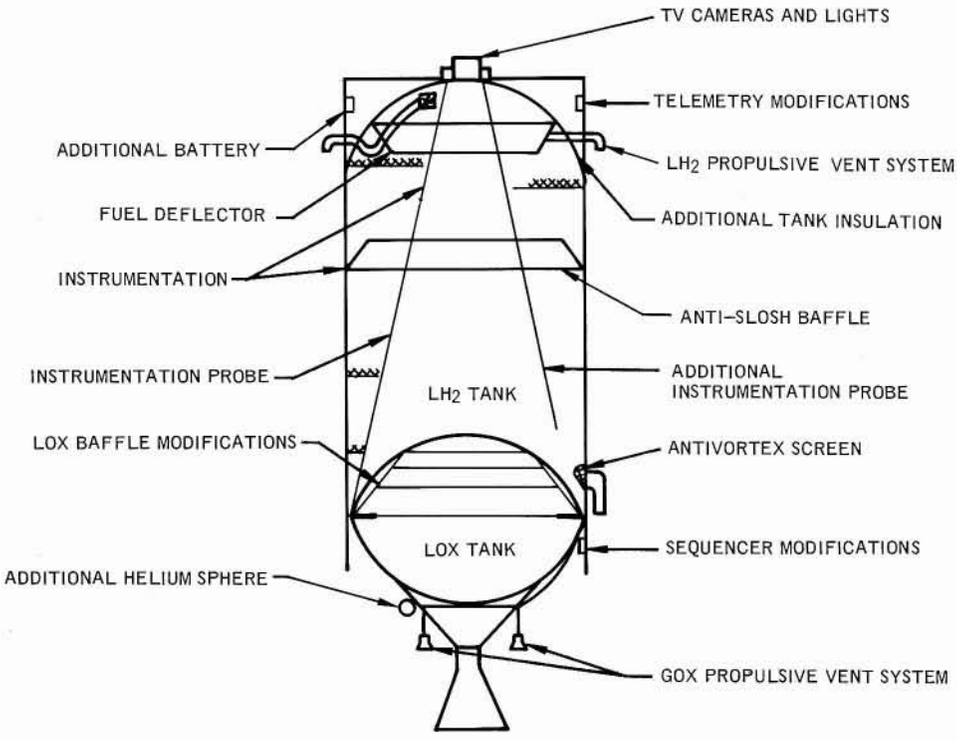
S-IVB STAGE MODIFICATIONS

ADDENDUM 1, FLIGHT VEHICLE AS-203



D-PB-500

MODIFICATIONS TO S-IVB FOR LIQUID HYDROGEN FUEL EXPERIMENTS (TO CONTROL BEHAVIOR OF FUEL IN SPACE UNDER LOW G CONDITION)

S-IVB CONFIGURATION

The S-IVB stage of the AS-203 has been configurated to simulate Saturn V S-IVB stage functions which include: orbital insertion and control of propellants during transition to low-gravity operation; orbital coast and LH_2 tank venting during three earth orbits; LH_2 tank repressurization and partial simulation of orbital restart.

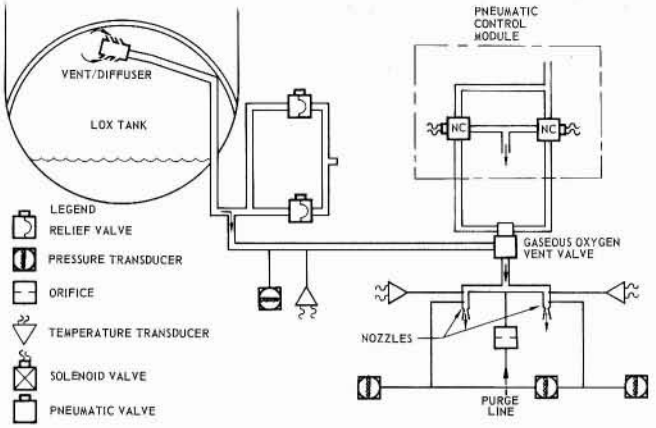
Modifications and additions to the stage include a repressurization system, hydrogen propulsive vent, gaseous oxygen (GOX) propulsive vent, LH_2 tank baffle and deflector, and provisions for special instrumentation to provide additional system control and additional data acquisition as required to satisfy the flight mission.

Differences between the S-IVB stage of AS-203 and a typical S-IVB stage are detailed by stage fabrication and assembly or stage systems.

STAGE FABRICATION AND ASSEMBLY

Fabrication and assembly of the S-IVB stage of the AS-203 differs from a typical stage as follows:

- A two-foot diameter helium sphere is mounted on the thrust structure to provide gaseous helium for repressurizing the LH_2 tank.
- The GOX propulsive vent assembly, consisting of aluminum ducts tapping off the LOX tank vent and relief valve, is installed on the aft dome of the propellant tank assembly. The ducts carry vented GOX to two aft-facing overboard nozzles which are mounted through the thrust structure, 180° apart.
- An anti-slosh baffle and a fuel deflector are installed in the LH_2 tank. The fuel deflector is installed approximately 10 feet from the forward end of the tank; the anti-slosh baffle is installed near the center of the tank. Both deflector and baffle are made of light nylon fabric, approximately two feet wide, and installed on the inside diameter of the tank wall.
- An antivortex screen is installed, covering the fuel line intake in the LH_2 tank. It is a half-cone of stainless steel mesh, fitted with a slotted top, which opens automatically to release bubbles that may form in the fuel line.
- An additional instrumentation probe assembly is installed in the LH_2 tank to support additional instrumentation requirements.



Gaseous Oxygen Propulsive Vent System

- Additional insulation consisting of aluminized mylar is installed on the outside of the LH_2 tank forward dome to provide increased thermal protection and to minimize hydrogen boil-off during the orbital coast period of flight.
- Two television cameras, with associated lighting equipment, and four quartz view ports are installed in the top of the forward dome of the LH_2 tank to visually relay LH_2 flight behavior pictures to monitoring stations.
- A propulsive vent system is installed on the LH_2 tank forward dome. It consists of aluminum ducts, a continuous vent control module, and two nozzles. The vent system ducts discharge hydrogen boil-off gases through the aft-facing nozzles mounted 180° apart through the forward skirt.

STAGE SYSTEMS

Modifications to the oxygen system, fuel system, pneumatic control system, electrical system, and telemetry and instrumentation system to accommodate the AS-203 mission are described in the following paragraphs.

Oxidizer System

The GOX propulsive vent system supplies approximately 30 pounds of thrust to the stage for positive ullage control, under low gravity conditions.

GOX enters the LOX tank vent/diffuser and flows to the GOX vent valve. On command from the Instrument Unit, a pneumatic control module opens the GOX vent valve, discharging GOX through two aft facing nozzles located on the thrust structure. Instrumentation is provided to monitor system operation.

In flight, the GOX propulsive vent opens immediately after J-2 engine cut-off, and remains open for approximately 77 seconds during the transition to low gravity flight.

Fuel System

Modification to the fuel system provides a fuel tank repressurization system and a LH_2 propulsive vent system.

FUEL TANK REPRESSURIZATION SYSTEM

A fuel tank repressurization system is installed on the stage to accommodate the simulated orbital restart experiment. The system utilizes gaseous helium at 3000 PSI from an additional storage sphere mounted on the thrust structure to repressurize the LH_2 tank prior to simulated J-2 engine restart.

The principal components of the system, in addition to those of a typical S-IVB stage, are the repressurization helium sphere, repressurization control module, and associated plumbing, valves, and controls.

Gaseous helium flows from the storage sphere through the repressurization control valves and a filter in the control module. It then enters the LH_2 tank through a diffuser in the forward dome.

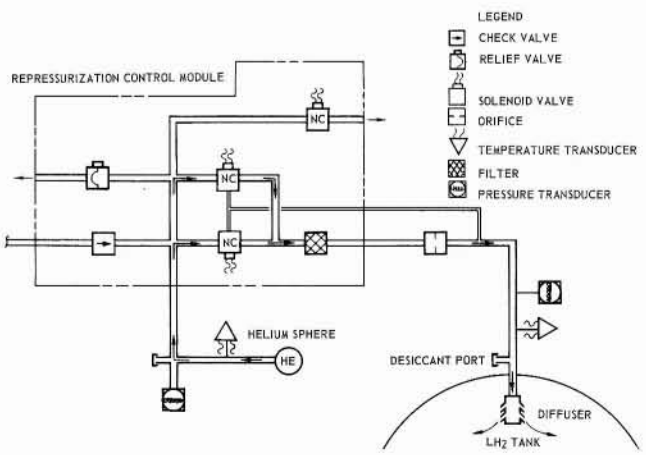
The repressurization control valves are opened on command from the Instrument Unit.

LIQUID HYDROGEN PROPULSIVE VENT SYSTEM

The LH_2 propulsive vent system provides approximately three pounds of continuous thrust during orbit to assist in ullage control. It consists of a propulsive continuous vent control module, associated controls, ducting, and two nozzles.

The system receives gaseous hydrogen boiloff from the LH_2 tank vent. It is routed through vent valves located in the continuous vent control module and discharged through two nozzles located on the forward skirt.

The propulsive continuous vent valves open on command from the Instrument Unit.



Fuel Tank Repressurization System

Pneumatic Control System

Modifications to the stage pneumatic system provides plumbing, pneumatically controlled valves, and switches to operate the GOX propulsive vent system, fuel tank repressurization system, LH_2 propulsive vent system, and television cameras and associated equipment.

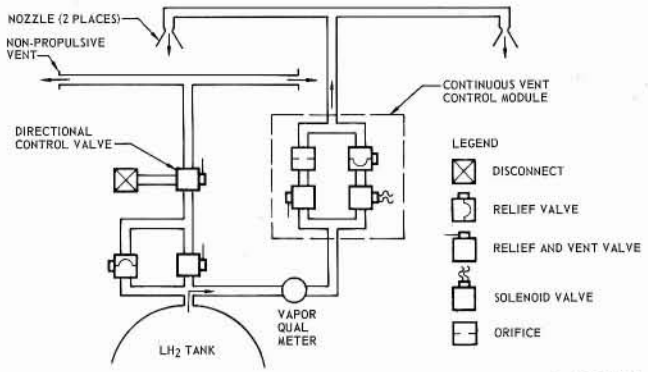
Electrical System

The electrical system has been modified by adding a 28 vdc battery to supply power for television cameras, camera heaters, and lights.

Telemetry and Instrumentation System

The telemetry and instrumentation system has been modified by adding hydrogen experiment oriented data collecting sensors together with the associated signal conditioning and transmission capabilities. The additional data collections are routed through the forward multiplexer assemblies to ground receiving stations.

Transmission of video signals from the two propellant tank mounted television cameras is routed through the Instrument Unit's existing transmitter system.



Liquid Hydrogen Propulsive Vent System